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FINAL STUDY REPORT

JUPITER ATMOSPHERIC ENTRY MISSION STUDY

Volume I

Management Summary

April 1971

MARTIN MARIETTA

DENVER DIVISION

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FINAL REPORT

Volume - Management Summary

April 1971

Approved

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FOREWORD

This report has been prepared in accordance with requirements of Contract JPL 952811 to present data and conclusions resulting from a six-month study effort performed for the Jet Propulsion Laboratory by the Martin Marietta Corporation, Denver Division. The report is divided into the following volumes:

Volume I - Management Summary;

Volume II - Mission and System Evolution;

Volume III - Supporting Technical Studies.

The report is arranged so that Volume I (Management Summary) will provide a concise overview of the study, Volume II (Mission and System Evolution) will provide an appreciation of the major mission and system integration and trade sensitivities, and Volume III (Supporting Technical Studies) will provide the detailed supporting tradeoff studies down to the subsystem level. Volume III also includes the appendixes with additional detailed data.

ACKNOWLEDGMENTS

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John M. Wilson	-	Mechanical Systems
Ludwig G. Wolfert	-	Science Mechanization

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I. INTRODUCTION

The material presented in this report summarizes efforts and products of a six-month study of a Jupiter atmospheric entry mission. Included are the study constraints, a summary of the science and mission objectives, a discussion of the mission analysis and selection rationale, a description of the required engineering implementation, and the conclusions and recommendations.

The basic study objectives were to assess the technical feasibility of a Jovian atmospheric entry mission in a 1978 Launch opportunity, and to define the gross mission and technology requirements. Probe survival to pressures up to 1000 atmospheres was to be evaluated and a realistic upper pressure limit selected that would permit the direct measurement of physical parameters and phenomena well below the tops of the visible clouds. This study is primarily concerned with questions representing the major scientific areas of interest in which investigation by a first generation entry probe is feasible. The science objectives for the mission were predefined in the form of 16 questions to be answered regarding the Jupiter atmosphere. These questions may be grouped into seven major categories: H/He ratio and isotopic abundances, composition and structure of the atmosphere, composition and structure of clouds, complex molecules, origin of colors, magnetic fields, and atmospheric turbulence.

The scope of the study included four major task areas: definition of science criteria, definition of planetary approach and entry trajectories, definition of entry probe systems, and definition of planetary vehicle system requirements. The science task included defining necessary instrumentation and the required science measurements and measurement intervals based on the science questions to be answered. The trajectory analyses task included consideration of targeting to meet the science objectives, approach and entry constraints for both direct and relay communications links, and entry and descent trajectory parameters affecting probe system design. The entry probe system task included consideration of propulsion system deflection design to achieve required targeting, data system design, including both direct and relay communications links, engineering subsystem definition, and evaluation of major tecnnology efforts necessary for the entry system. The planetary vehicle system task was limited to identifying the interface requirements and spacecraft modifications necessary to support the entry mission within constraints typical of the TOPS and the Pioneer F/G spacecraft.

The study was accomplished in three major phases: (1) Trial Mission Definition; (2) Parametric Analyses and Tradeoff Studies; and (3) Mission Definition.

The Trial Mission Definition phase included definition and synthesis of a representative mission in order to focus the parametric analyses and tradeoff efforts toward pertinent ranges of study and to provide a method of uncovering critical problem areas. The effort was intended to demonstrate a preliminary implementation of the engineering subsystem design and integration.

The parametric analyses and tradeoff studies supported both the trial mission synthesis and the establishment of the baseline missions, while the last phase included definition of specific missions and variation of the primary mission parameters so that the effects of these variables on the overall entry system design could be evaluated. In addition, sample missions were defined to demonstrate some unique types of missions of particular interest.

The services of a group of interested planetary scientists were solicited to help interpret the scientific objectives in terms of relevant measurements and requirements compatible with a descent probe and to comment on the scientific adequacy of the mission designs. The group consisted of Drs. S. I. Rasool (NASA/GSFC), D. M. Hunten (KPNO), T. Owen (State U of NY), C. Sagan (Cornell), and R. Goody (Harvard). Dr. Rasool participated as a NASA reviewer. Scientific representation from JPL included Drs. R. Newburn, R. A. Schorn, and W. S. McDonald.

Technical advice was obtained from various individuals and companies in support of the engineering systems definitions. Johns Manville Company of Manville, New Jersey, provided data on the Min-K insulation for probe thermal protection, and Royal Industries of Santa Ana, California, provided data on phase change materials. In the development of estimation procedures for the microwave atmospheric attenuation model, the following individuals provided valuable consultation: Dr. A. A. Maryott of NBS, Washington, D. C.; Dr. Roger Gallet of NBS, Boulder, Colorado; and James Gallagher of Georgia Institute of Technology. Useful design data were obtained from personnel of Ball Brothers Research Corporation, Boulder, Colorado, in the area of despun antennas for application to the spinning Pioneer F/G. Ames Research Center personnel provided further details on the Pioneer F/G spacecraft.

II. SUMMARY

The objective of this study was to investigate the feasibility of a Jovian atmospheric entry mission with probe survival to a pressure of 1000 atmospheres and to recommend a realistic upper pressure limit that will allow direct measurement of physical parameters and phenomena well below the tops of the visible clouds. A major uncertainty, the entry survival and heat shield performance, was not a part of this study and, in order to limit the scope of the effort, the heat shield weight fractions were supplied by JPL. The Titan IIID/Centaur/Burner II family of launch vehicles with launch payload up to 2400 lb was considered. Although the major effort centered around a 1978 single launch, other missions and launch years were investigated. Both TOPS and Pioneer F/G spacecraft were considered and requirements for modification were identified for each.

The overall mission engineering feasibility centers around the science objectives, obtaining the necessary measurements within the environment, and returning the data. The services of a group of interested planetary scientists were solicited to help interpret the scientific objectives in terms of relevant measurements and requirements compatible with an atmospheric descent probe. Although the study ground rules specified a nominal science payload (which included a gas chromatograph/mass spectrometer, temperature and pressure sensors, accelerometers, and visual photometers), an expanded and also a reduced payload were designed. These were based on the recommendations of the planetary science consultant group and results of the project studies.

Figure II-1 presents the basic study flow logic used to transform the science questions and resultant mission objectives into the final engineering hardware requirements of the total system. The report, as well as the study, essentially follows this pattern. From the science objectives, both the science instrument implementation and the overall mission objectives were established. The mission analysis and selection of practical missions follows. Consideration was given to combining the probe mission with various Grand Tour missions to take advantage of program cost savings. In addition, an interesting mission was conceived that could deliver two small probes, one to Jupiter and the second to a subsequent planet on a Grand Tour trajectory.

In the engineering design of the system, the critical areas included telecommunications including the probe/spacecraft geometry required for a relay link, probe survival and the thermal/structural design to 1000 atm pressure, heat shield/aeroshell weight penalties incurred at higher entry angles, and accuracy requirements for the deflection-to-entry maneuver. Design solutions to all of these engineering problems are feasible, and a broad range of missions are available using a spacecraft relay link.

The two major study uncertainties that may have the most effect on the mission performance are the unknown structure and composition of the Jupiter atmosphere, and the uncertainty in the prediction of the heat shield performance. Large variations in these factors can be accommodated in feasible system designs; however, some penalties in performance must be accepted when designing to the extremes.

The following sections present the mission study characteristics and ground rules, the science prospectus, the mission analysis and selection, and the results of the engineering implementation.

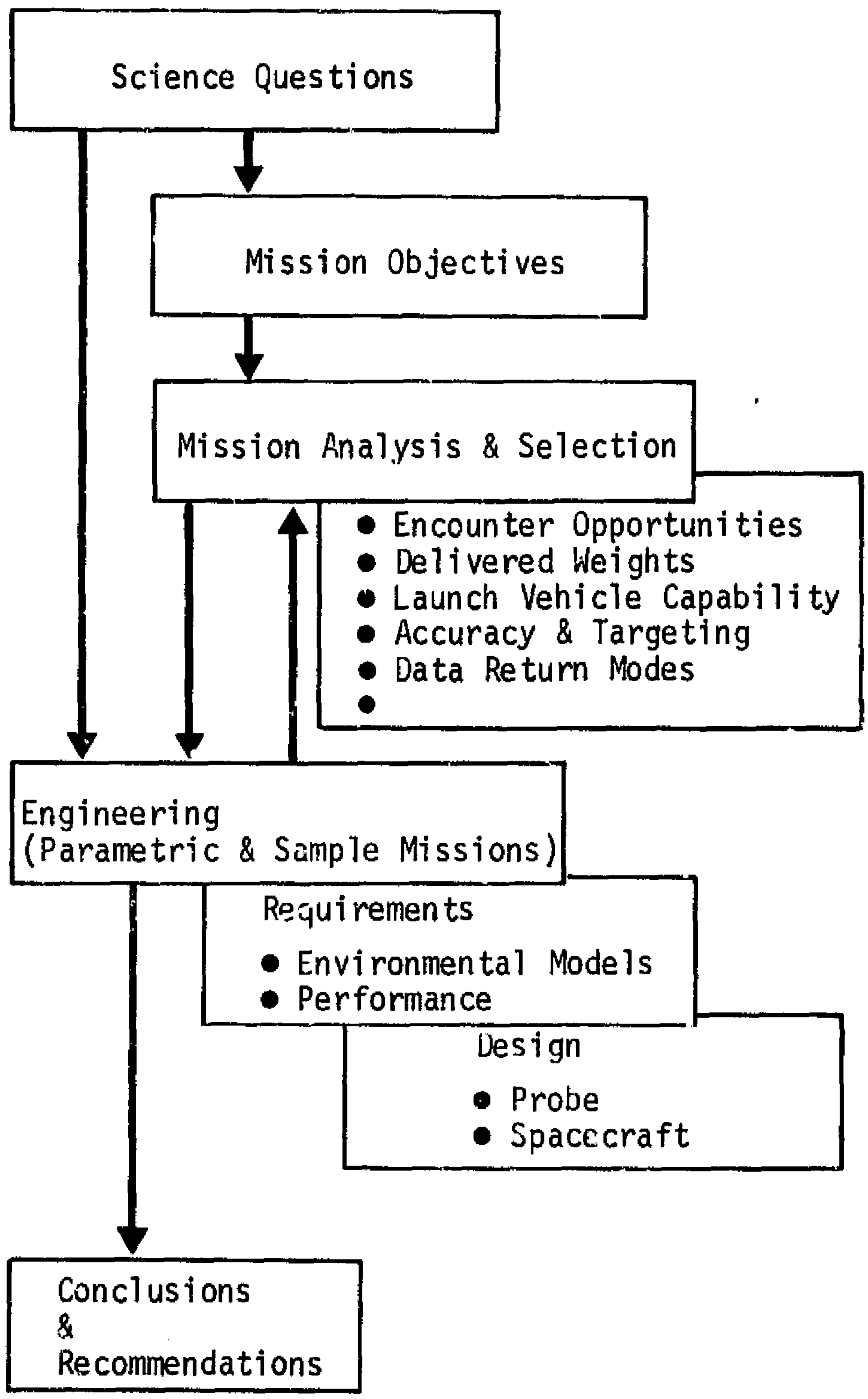


Fig. II-1 Study Flow Logic

A. MISSION STUDY CHARACTERISTICS

This section identifies the basic scientific and engineering questions to be answered in a first-generation Jupiter atmospheric probe mission, presents the study assumptions and ground rules, and summarizes the system performance expectations required to provide a feasible mission.

1. Science Questions

The major scientific areas of interest for a first generation probe to the lower atmosphere of Jupiter are represented by the set of basic questions shown in Table II-1. The determination of the H/He ratio for the outer planets, particularly on Jupiter, is generally accepted as the most valuable information that could be learned from a space probe because of its importance to questions of the origin of the solar system and cosmology. This quantity cannot be determined unambiguously in any other fashion than by *in situ* measurements from a probe that survives entry and descends below the turbopause to the lower atmosphere. The relative abundances and isotopic ratios for the other elements up through argon are also of significance to the same questions and also require a descent to the lower atmosphere. Abundances and isotopic ratios for Li. Be, and B would set an upper limit on Jupiter's past temperatures. However, while the H/He ratio, and the abundances and isotopic ratios of most other important elements up to mass 40 are expected to be measurable in the low pressure ($p \leq 1$ atm) regions of the atmosphere, Li, Be, and B are not expected to be present in measurable quantities much above the 10,000-atmosphere level.

The results of this study have shown that at least partial answers to all of the questions listed in Table II-1 can be obtained from survivable entry probes. The degree of completeness in the answers will depend strongly on the depth and nature of the descent below the 1-atmosphere level and on the amount and sophistication of instrumentation that can be carried. The study reported herein documents the relative performance and feasibility of obtaining the various degrees of completeness in the answers.

Table II-1 Basic Scientific Questions for Jupiter (JPL Section Document 131-07)

1.	What are the relative abundances of hydrogen, deuterium, helium, neon, and other elements, and what are their isotopic compositions?
2.	What are the present-day atmospheric composition and altitude profiles of pressure, temperature, and density, and what effect do they have on the radiation balance?
3.	What are the chemical composition and vertical distribution of the clouds?
4.	Do complex molecules exist in the atmosphere of Jupiter?
5.	What are the nature and origin of the colors observed in Jupiter's atmosphere?
6.	What is the magnetic field strength in the lower atmosphere?*
7.	What is the level of turbulence in the atmosphere?*
*Supplementary questions added after start of contract.	

2. Engineering Questions

For a first-generation Jupiter entry mission, the engineering design margins must be large to overcome the uncertainties in the environment and the performance uncertainties of engineering systems that may have only been partially tested due to an inability to duplicate the flight conditions in test. Probe systems flown on subsequent missions can be designed more efficiently and reliably if the first flight probe obtains answers to basic engineering questions.

The engineering questions to be answered fall into two categories. The first category includes measurements made to determine how well the engineering systems performed, and the second category includes measurements made to determine specific characteristics of the Jupiter environment that may have an impact on the engineering system design.

The performance evaluation instrumentation will be used to determine system status from prelaunch to the end of the mission; however, its most important functions are performed while the probe is actively performing its entry mission. The heat shield performance can be evaluated with temperature and heat shield recession histories and the pressure vessel/thermal control subsystem requires pressure and temperature measurements to evaluate insulation performance as well as structural and leak integrity. Critical voltage levels are required within the power, communications, sequencer, and data systems. Jupiter atmospheric attenuation can be evaluated by measurement of received signal strength at the spacecraft.

Engineering experiments on specific characteristics of the Jupiter environment were not considered in detail during this study; however, it is likely that follow-on studies will require consideration of this category. The following list might be considered:

Entry heating rate peaks;
Wind and turbulence measurements;
Radiation belt energy distribution.

Direct measurement of entry heating rate peaks would be extremely difficult, if not impractical, although some estimates might be inferred from recession and temperature data. Accurate wind and turbulence information would require accelerometer data rates higher than those provided in this study. Extreme local turbulence could result in loss of the data link due to large excursions in probe antenna pointing.

The measurement of the magnitude and distribution of the radiation belt from the entry probe would provide both engineering and scientific data. Estimates of the large uncertainties in this model result in some shielding weight penalties.

3. Study Assumptions

The following assumptions or guidelines were used to bound the study:

1) Mission Accomplishment Shall Be During the 1978 Launch Opportunity (specified by JPL) - The baseline mission studies used the 1978 launch opportunity, however, a 1979 Jupiter-Uranus-Neptune (JUN) Grand Tour launch opportunity was also included to determine the effects on the probe systems of particular trajectory constraints associated with that mission.

2) System State-of-the-Art Will Be as of July 1975 (JPL furnished constraint).

3) <u>Baseline Science Payload</u> - The baseline or nominal science payload was specified by JPL. It included an ion mass spectrometer, a gas chromatograph/neutral mass spectrometer, six photometer channels, and an aerometry package containing accelerometers and pressure and temperature sensors. The baseline payload was used for the parametric mission studies and the design example mission. However, other instrumental techniques and payloads appropriate to the stated objectives were also investigated and two of the sample missions discussed in Volume II incorporate payloads with expanded and contracted capabilities.

4) <u>Atmospheric Model</u> - The preliminary model atmospheres were furnished by JPL and were based on an in-process NASA Design Criteria Monograph (Ref II-1). Since the study began, two improved sets of models dated 8 May 1970 (Ref II-2) and 8 October 1970 (Ref II-3) have been circulated. The nominal model in all three documents remained essentially the same and was used as the baseline criterion for all analyses and system designs. However, the effects of encountering other plausible models were investigated. One example mission was designed to survive in either the nominal or the cool models of Reference II-2, shown in Fig. II-2. The warm model is considered much less plausible than the other two and was not used in the study. The models are described more completely in Volume II, Chapter II.

5) <u>Trapped Radiation Model</u> - The trapped radiation belt model used for the study was furnished by JPL and was based on an in-process NASA Design Criteria Monograph.

The detailed model (Ref II-4) describes for relativistic electrons, energies of the order of 10 Mev and peak fluxes of the order of 10^7 cm^{-2} sec^{-1} at about two planet radii from the dipole.

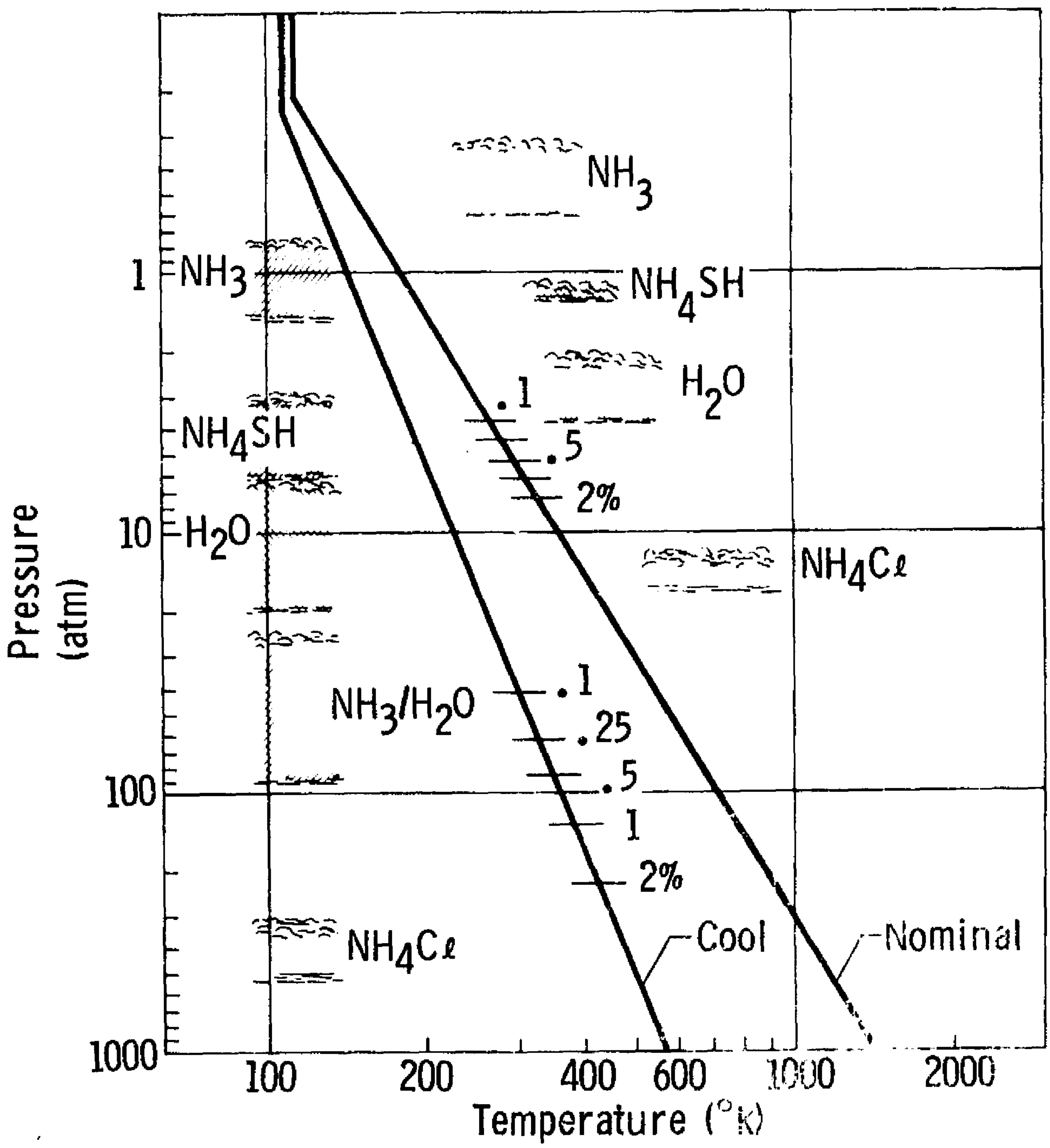


Fig. II-2 Cool-Nominal Model Atmospheres

An engineering model describing the distributions of relativistic electrons has been developed, including uncertainty estimates. Models have also been developed for energetic protons and for low-energy particles, but these models are not securely based on observational data. The uncertainties associated with these models are very broad, and could be reduced only by considerable study and/or direct detection by spacecraft. Although the worst-case model implies a very severe radiation environment, the nominal case model is well within current design limits. Further discussion is in Volume II, Chapter II.B.

6) Micrometeoroid Model - The micrometeoroid model used in this study was assumed by Martin Marietta based on References II-5 and II-6. The risk of penetration by a micrometeoroid of significant size ($\sim$0.01 gm) on a Jupiter mission can be considered to be the sum of the risks associated with the near-earth region, the interplanetary region, and the region of the asteroid belt. It is shown by Volkoff (Ref II-5) that on a Jupiter mission the risk per unit time in the near-earth region is at least an order of magnitude greater than in the asteroid belt and several orders of magnitude greater than in other regions.

Protective requirements are discussed in Volume II, Chapter II.A.

7) Launch Vehicle Performance - The launch vehicle is a member of the Titan III/Centaur family as defined by JPL (Ref II-7). Titan III/Centaur family launch load performance data are presented in Fig. II-3. In addition, a stretched version of the Centaur stage has been considered in the parametric studies and its impact is presented in Volume III, Chapter III. Considering the shorter required launch period for 5-segment vehicles, and therefore lower C_3 values required for a given mission, the payload capability with the

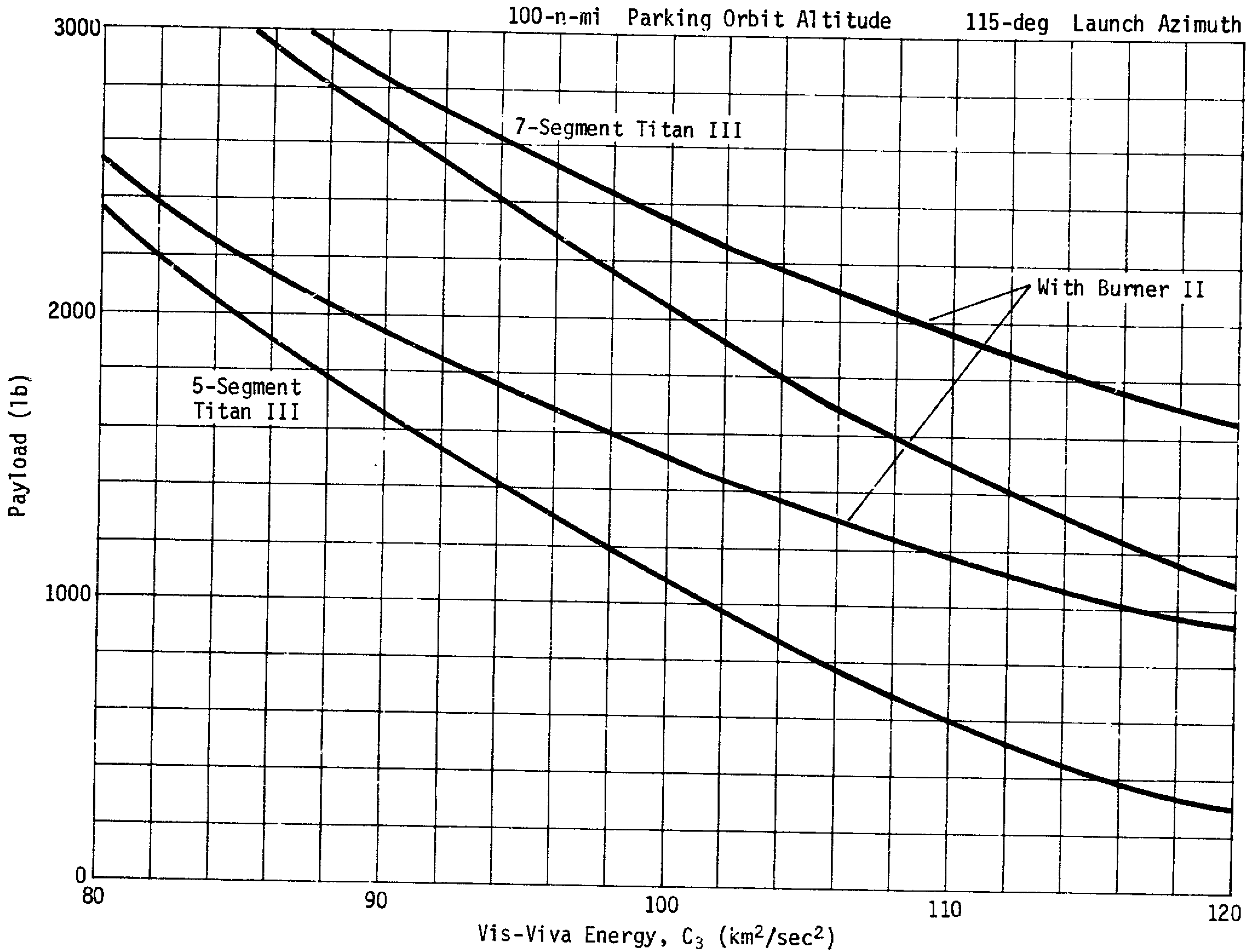


Fig. II-3 Titan III/Centaur Load Performance Data stretched Centaur is nearly equivalent to the basic 7-segment vehicle.

The allowable parking orbit coast time in Reference II-7 is 0.5 hr or less. This constraint eliminates Type I trajectories from consideration in 1978. However, refinements already considered for Centaur make this a rather "soft" constraint and it has been extended to 1 hr for this study. By extending the maximum allowable coast time to 1 hr, this parameter no longer constrains the mission design in any way.

Minimum launch periods of 20 days for the Titan IIID/5-segment vehicle and 30 days for the Titan IIID/7-segment vehicle are specified in the reference document. No relaxation of this constraint was considered at this point in the study. However, the cost or benefits of other launch period lengths is shown in the parametric data.

The launch azimuths have been limited to 90 to 115 deg and result in declinations of the launch asymptote (DLA) of less than ±36 deg. Conversations with JPL personnel led to the consideration of northerly launch azimuths that will yield DLA values up to 40 deg. The result of these variations is discussed in Volume III, Chapter III. In summary, the effect of DLA constraints other than ±36 deg does not significantly affect the mission capabilities for the range of variations that can be reasonably considered.

DLAs of less than ±2 deg are not considered because of tracking considerations for the early midcourse correction maneuvers. This constraint might be removed or relaxed with further study. However, the minimum DLA constraint does not significantly affect the mission design.

8) Astronomical Constants - Astronomical constants were furnished by JPL (Ref II-8) and summarized in Table II-2. Discussion of their use is included in Volume III, Chapter III. Zero altitude is assumed to be at a radius of 71,420 km (nominal radius yielding pressure of 1 atmosphere).

9) Transfer Trajectory Data - Data for the 1978 mission opportunities has been provided by JPL in tabulated form. Figure II-4 presents the launch and arrival date combinations and energy, C_3, requirements. The resulting hyperbolic excess velocities at Jupiter, V_{HP}, are noted along with the DLA constraints from Item 7) above. Similarly, good communications from the planet back to earth require that the earth-spacecraft line of sight be at least 15 deg from the earth-sun line of sight. This constraint is also noted in the figure.

Other launch year data (1979, 1981) have been supplied and are discussed in Volume III, Chapter III. Other mission types such as 1976 and 1977 Jupiter-Saturn-Pluto multiple planet flybys and 1978, 1979, and 1980 Jupiter-Uranus-Neptune Grand Tours are also discussed in the parametric data.

10) TOPS and Pioneer F/G Spacecraft Interfaces - TOPS and Pioneer F/G spacecraft descriptions were provided by JPL (Ref II-9) for use in the study as examples of realistic spacecraft characteristics. The two most important differences between the spacecraft when used in conjunction with the Jupiter probe mission are their total weight and mode of stabilization. The unmodified Pioneer F/G weighs 547 lb and the unmodified TOPS weighs 1450 lb, giving the Pioneer F/G a sizeable weight advantage. The TOPS spacecraft is three-axis stabilized while the Pioneer is spin-stabilized. For missions where the spacecraft provides a data relay mode with probe tracking antenna, the TOPS provides a much simpler system.

Table II-2 Astronomical Constants

The constants used to define the ephemerides of the planets Jupiter and Earth were obtained from JPL TR 32-1306, *Constants and Related Information for Astrodynamic Calculations, 1968.* The mean ecliptic elements that define the ephemerides of the planets are shown below.

	Jupiter	Earth
a = semimajor axis ~ A.U.	5.202803	1.00000023
i = inclination of orbit to ecliptic	1° 18' 31".3 - 20".0 * T	0.0
Ω = longitude of ascending node of orbit on ecliptic	99° 26' 16".3 + 3639".5 * T	0.0
$\tilde{\omega}$ = longitude of perihelion	12° 42' 41".12 + 5800".79 * T	101° 13' 15".0 + 6189".03 * T + 1".63 * T^2 + 0".012 * T^3
e = eccentricity	0.0483376 + 0.00016302 * T	0.01675104 - .418 x 10^{-4} * T - .126 x 10^{-6} * T^2
M = mean anomaly	225° 13' 17".7 + 299".123557 * d	358° 28' 33".04 + 1295 96579".1 * T - 0".54 * T^2 - 0".012 * T^3

d is the number of days from the epoch of 1900 January 0.5 ET and T is the number of Julian centuries of 365.25 days from the same epoch.

The orientation of Jupiter's rotational axis, with respect to the mean earth equator and equinox of date, is defined by

α = right ascension of pole = 268°.0035 + 0.00103 (t - 1910.0)

δ = declination of pole = 64°.5596 - 0.00017 (t - 1910.0)

Other pertinent constants for Jupiter include

μ = gravitation constant = 1.267077188 x 10^8 km^3/sec^2
= .44746367 x 10^{19} ft^3/sec^2

P = rotational period = 9.841667 hr

ω = rotational rate = 1.7734881 x 10^{-4} rad/sec

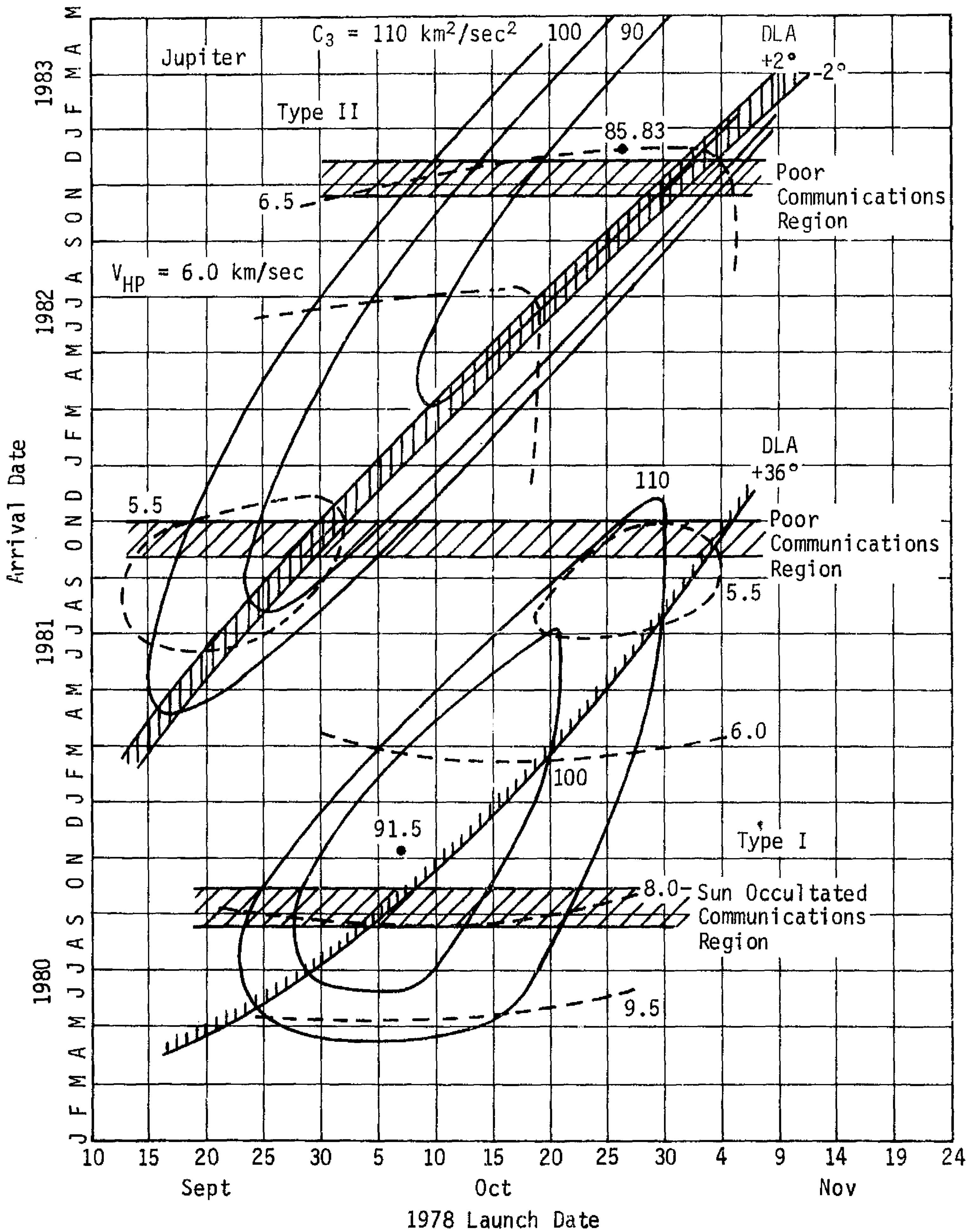


Fig. II-4 Launch and Arrival Date Combinations

11) DSN Capability - DSN capabilities were furnished by JPL based on Mark III and IV projections (Ref II-10 and II-11). The spacecraft-to-DSN links were not studied on this program because this was outside the scope of the contract. The direct probe-to-DSN link used on one mission studied and the DSN performance contributions to the navigation and accuracy analysis were analyzed in detail. It was found that the capabilities specified in Reference II-10 were adequate to carry out the Jupiter atmospheric probe missions. Improved navigation accuracy would be desirable for multiple-planet missions, but probe deflection errors dominate navigation errors in the Jupiter missions treated in this study.

12) Heat Shield Performance - Quoting the JPL furnished data:

> "Jupiter atmospheric entry heating rate and pressure histories are for the most part outside our present experience. Because of this, the normal analytical tools tend to be very limited in application and where extrapolations are possible they tend to be unverifiable through test. For this reason, a wide variety of opinions are possible as to the heat shielding weight requirements for typical Jupiter missions. In order to constrain these studies to one tier of speculation, JPL has furnished heat shield weight fraction data to guide heat shield weight assignment for different probe missions."

These data were based on the work done for this study by Tauber and Wakefield at Ames Research Center. Their data are reproduced for reference in Volume III.

13) Planetary Quarantine - Paragraph I (b) (4) (D) of the JPL Statement of Work is quoted below:

> "Planetary quarantine should not be considered in this study, although NASA document NHB 8020.12, 'Planetary Quarantine Provisions for Unmanned Planetary Missions,' dated April 1969, is expected to apply to a specific mission."

In accordance with the quoted direction no design requirements for compliance with NHB 8020.12 have been considered. Nevertheless the provisions for planetary quarantine must be considered during the early phases of mission selection, planning, and definition of system requirements.

During design of each subsystem, consideration must be given to materials selection and sterilization procedures. For the descent probe, insulated to withstand heat, either extended heat soak periods or unique sterilization procedures will be necessary.

Spacecraft trajectories near the target planet will determine allocation of contamination probability for the spacecraft. For 1979 JUN Grand Tour missions with a large flyby radius of 6.8 R_J, the spacecraft contamination control may be relaxed over that of direct impact or near flyby trajectories considered for the 1978 missions. However, weight penalties will be incurred for such hardware as biocanisters and other provisions for contamination control between the launch vehicle, spacecraft, and entry probe.

4. System Performance Expectations

The mission feasibility is dependent on accurate delivery of the probe to the target, probe survival through both entry and descent, measurement and return of the science data, and overall system reliability to endure the long mission times. The following paragraphs discuss the general level of system performance required to provide feasible missions.

a. Navigation and Probe Deflection Systems - The navigation system performance characteristics during the approach phase are evaluated assuming Deep Space Network (DSN) doppler tracking only and DSN doppler data coupled with Jupiter/Canopus angle measurements made by a sensor onboard the spacecraft. The spacecraft orbit determination uncertainties at probe deflection time are calculated assuming optimal data processing for a range of DSN tracking station location errors* and onboard sensor measurement errors.† All navigation data are based on Jupiter ephemeris errors of 500 km (1σ), which are in agreement with ephemeris accuracy predictions (Ref II-12) for the late 1970's.

The effects of probe deflection maneuver implementation errors are evaluated for accuracies ranging from those expected (Ref II-13) for the Viking deorbit system -- 1% (3σ) proportionality and 1 deg (3σ) pointing -- to errors four times larger. Deflection accuracies of 1% and 1 to 1.5 deg should be realizable for a probe deflected from the TOPS vehicle. Somewhat larger errors can probably be expected for probe deflection from the spinning Pioneer vehicle. A detailed error analysis is required to determine these errors but they should be well within the range considered in this study.

*Errors in position relative to earth's spin axis = 0.5 to 2.5 m; longitude errors of 1 to 5 m.

†Errors from 5 to 50 arc-sec.

The probe deflection maneuver errors dominate the entry dispersions and navigation errors may be neglected for Jupiter atmospheric probe missions. As a result, onboard navigation sensors will not be of value for these classes of missions. The deflection maneuver implementation accuracy is a key factor in the relay communications link design.

b. Survival - If the science and engineering payload can be delivered on target to Jupiter and provisions for telecommunication of data through the dense atmosphere can be made, there remains the task of protecting the payload from the high velocity entry and the subsequent slow descent into extreme pressure and temperature levels. The entry survival is not the subject of this study; sufficient evidence that entry is feasible has been found in studies by JPL and Ames Research Center and the weight fractions necessary to accomplish it have been assessed. Therefore the effort in this study is devoted to the area of descent survival. To that end, existing structural alloys and insulation materials are evaluated to establish their applicability at temperatures up to 2100°F and pressures of 1000 atm. With the performance estimated from these materials, the weight fraction of the portion of the postentry descent probe required to provide environmental protection has been found to range from 20%, for probe depths of 50 atm and descent times of 1 hr, to 60% for descent depths of 1000 atm and times of over 2 to 3 hr. These performance levels, in conjunction with the entry protection weight fraction values of 30 to 40%, result in adequate probe mission performance.

c. Data Transfer System - The rate of data transfer is dependent on the depth of descent of the probe into the Jupiter atmosphere. Probe transmitter power requirements range from 10 to 40 watts for the missions studied. Modulation by convolutional encoding in a PCM/PSK/PM system gives from 360,000 to 648,000 bits of preentry data, and from 144,000 to 273,600 bits of data for the entry and postentry phases. A typical data rate of 180 bps is used during the preentry phase, with switching to 40 bps for the initial postentry phase, and switching to 20 bps for depths lower than 30 atm. Extreme depths (1000 atm) require data rates of 10 bps. Relay link frequencies range from 0.85 GHz to 2.3 GHz for the different missions.

The spacecraft mounted receiving system performs search and acquisition to acquire the probe signal then autotracks in frequency and in antenna pointing. Tracking loop bandwidths of 30 Hz to 100 Hz and search and acquisition times of approximately two minutes are anticipated.

Direct links to Earth (without spacecraft relay) can provide 20 bps but these systems are costly in weight and are restricted in targeting, descent time, and depth by the line-of-sight constraint.

d. Reliability - A cursory evaluation of the reliability requirements for a Jupiter atmospheric probe can be made by comparing the probe systems and anticipated environments to those of well-defined systems. Current and proposed spacecraft systems encompass the mission duration required. As a general comparison of long life mission design, the following categories are representative:

Spacecraft	Approximate Design Lifetime (year)
Mariner	2
Pioneer	2 to 5
TOPS	5 to 12

The Jupiter atmospheric probe mission duration may vary from about 2 years for the Type I trajectories, to about 3 years for the Type II trajectories. This would imply reliability requirements typical of the Pioneer spacecraft and, therefore, state-of-the-art hardware would, in some instances, be sufficient.

B. SCIENCE PROSPECTUS

1. Scientific Objectives and Observables

The set of basic questions shown in Table II-1* were interpreted in terms of relevant measurements or observables compatible with the descent probe concept and limited to information that can be obtained only by *in situ* measurements. This interpretation of the questions in terms of observables was aided by the comments and suggestions of the group of consulting scientists. The group consisted of Drs. S. I. Rasool (NASA/GSFC), D. M. Hunten (KPNO), T. Owen (State U of NY), C. Sagan (Cornell), and R. Goody (Harvard). Dr. Rasool participated as a NASA reviewer. Scientific representation from JPL included Drs. R. Newburn, R. A. Schorn, and W. S. McDonald. Table II-3 lists the resulting set of observables that served as specific scientific objectives for this study.

2. Measurement Performance Requirements

To use the observables as mission design and evaluation criteria, the specific conditions that must be met for the measurements to be properly relevant were established, again with the aid of the consultants. These performance requirements include the desired entry locations or targets, the pressures (depths), and the altitude sampling intervals appropriate to each observable.

The requirements, summarized in Table II-4, are based on the wide range of present theories and speculation on what may be found below the cloud tops, and are therefore somewhat subjective. This present state of ignorance will most likely exist until a probe actually enters and survives to a reasonable depth, i.e., 100 atm.

*See page II-5.

Table II-3 Relevant Measurements or Observables Derived from Basic Questions

1. Determine the relative abundances of H and He in the lower atmosphere (below the turbopause).
2. Determine the isotopic ratios H/D, $^{3}He/^{4}He$, $^{20}Ne/^{23}Ne$, $^{36}A/^{40}A$, $^{12}C/^{13}C$ in the lower atmosphere (below the turbopause).
3. Determine the atmospheric mean molecular weight and identify the major contributing gases (i.e., determine whether H_2 and He are indeed the only major constituents and, if not, what the other gases are).
4. Determine the concentration profiles (versus pressure and temperature) of the minor atmospheric gases (e.g., Ne, A, N_2, etc and CH_4, NH_3, H_2O, H_2S, etc) from above the visible cloud tops down to several hundred atmospheres.
5. Determine the temperature versus pressure (and time) profile from above the cloud tops down to well below the condensation level of H_2O with a precision sufficient to determine whether the lapse rate is adiabatic.
6. Determine the vertical distribution and structure of the cloud layers with respect to pressure and temperature (particularly, locate the cloud tops).
7. Determine the chemical composition of the cloud particles in each layer.
8. Determine the color of each of the cloud layers.
9. Determine the intensity distribution of the incoming solar flux (direct and diffuse) at several wavelengths as a function of pressure and temperature from above the visible clouds down to at least several tens of atmospheres.
10. Determine the thermal radiation (IR) flux profiles at several wavelengths from above the cloud tops down to several hundred atmospheres.
11. Determine whether specific complex molecules are present in the region between the cloud tops and the condensation level of H_2O.
12. Determine the frequency of occurrence of electric discharges and the nature of any thunder as a function of pressure and temperature down to at least the condensation level of H_2O.
13. Determine the physical characteristics (number density and size distribution) of the cloud particles in each layer (particularly through the cloud tops).
14. Determine the scales and the magnitude and frequency spectra of any atmospheric turbulence from above the cloud tops down to at least several tens of atmospheres.
15. Determine the magnetic field strength and variations versus depth from above the ionosphere down through the lower atmosphere to as deep as possible.
16. Determine the electric conductivity of the deep atmosphere.
17. Determine the relative abundances and isotopic ratios of Li, Be, B.
18. Determine the composition profiles of the ionic species through the upper atmosphere.
19. Determine the exospheric ionospheric temperature profiles.
20. Locate the source of decametric radiation (with respect to radius).

Table II-4 Summary of Performance Requirements for Observables

Observable	Targets	Pressure Depth	Sampling
1. H/He Ratio	Any	Below Turbopause	At Least 4
2. Isotopic Ratios	Any	Below Turbopause	At Least 4
3. Mean Molecular Weight	Any	To 5 atm or More	At Least 4
4. Minor Constituents	Any	To 100 atm or More	2 to 5 per Scale Height
5. Temperature/Pressure	EZ, NEB, TR, Poles	To 1,000 atm	50 to 100 per Scale Height
6. Cloud Layering	EZ, NEB, TR, Poles	To 100 atm or More	100 per Scale Height
7. Cloud Composition	EZ, NEB, TR, Poles	To 100 atm or More	2 to 5 per Scale Height
8. Cloud Colors	EZ, NEB, TR, Poles	To 100 atm or More	100 per Scale Height
9. Solar Flux	Subsolar or LS	To 10 atm	100 per Scale Height
10. IR Flux	Any	To 100 atm or More	50 to 100 per Scale Height
11. Complex Molecules	GRS, Any	To H_2O Cloud (5 to 100 atm)	2 to 5 per Scale Height
12. Lightning/Thunder	Any	To H_2O Cloud (5 to 100 atm)	10 to 20 per Scale Height
13. Cloud Particle Sizes	EZ, NEB, TR, Poles	To 100 atm or More	100 per Scale Height
14. Turbulence	EZ, NEB, Poles	To H_2O Cloud (5 to 100 atm)	10 to 20 per Scale Height
15. Magnetic Fields	Poles, GRS, or Any	To 1,000 atm or More	2 per Scale Height
16. Conductivity	Poles, GRS, or Any	To 1,000 atm or More	2 per Scale Height
17. Li, Be, B Ratios	Any	10,000 atm	At Least 4
18. Ionosphere Composition	Subsolar, LS, DS	Preentry	2 per Scale Height
19. Upper Atm Temperature	LS, DS	Preentry	2 per Scale Height
20. Decameter Radiation	Any	To 100 atm or More	---

*Targets: EZ = Equatorial Zone GRS = Great Red Spot NEB = North Equatorial Belt
LS = Lightside TR = North or South Temperate Regions DA = Darkside
Any = Any Target except GRS

There are four identifiable depths or pressure levels through which measurements must be made to satisfy the requirements of the various observables. These levels are: (1) the levels below the turbopause but above the clouds ($P \leq 1$ atm) where the H/He ratio and the gross atmospheric composition can be determined; (2) the 1 to 10 atmosphere levels to obtain at least some minimal information on the pressure-temperature profiles and the clouds; (3) the 100 to 300 atmosphere levels to ensure reaching below the condensation level of H_2O and to search for complex molecules; and (4) the 500 to 1000 atmosphere levels to satisfy curiosity. While descent to the 100 to 300 atmosphere levels would satisfy the requirements of most of the observables, descent to the 1000 atmosphere levels would be of an exploratory nature and is not strictly required by the objectives of a first-generation mission.

The sampling interval requirements also fall into several general categories. These are: (1) those that require only a few (four for redundancy) measurements anywhere in the mixed lower atmosphere such as the H/He and isotope ratios or the mean molecular mass; (2) those that require a few (two to five) measurements per scale height, such as the gross pressure-temperature structure, or the cloud composition and minor constituent profiles; (3) those that require averaging or integrating over an interval such as average turbulence or lightning measurements; (4) those that require 50 to 100 measurements per scale height, such as the detailed thermal and turbulence structure; and (5) those that require very detailed profiles (100 to 200 per scale height), such as the cloud structure and physical properties measurements.

Many of the sampling requirements could be relaxed and still give useful information. For example, a detailed pressure-temperature profile down to the 100 atm level combined with a few precise composition measurements near 100 atm might allow an inference of the cloud layering above the 100 atm level.

A probe entry point within 30° of the subsolar point in either the North Equatorial Belt or zone is optimum for all observables. However, entry within 50° to 60° of subsolar will still give some information on the vertical distribution of the solar flux. A darkside entry point would preclude such measurements but would not degrade the accomplishment of the other objectives.

3. Instrumental Techniques and Options

The major part of the study was concerned with the definition of probe system capabilities for carrying the baseline payload to various depths in the nominal model atmosphere. In addition, other instrumental techniques appropriate to the observables were investigated. Table II-5 summarizes the options or instruments for each of the observables.

4. Payload Selection and Description Summary

Several payloads in addition to the nominal were defined to evaluate a spectrum of missions with expanded and contracted science capabilities. The payloads ranged from a minimum complement of pressure and temperature sensors, a nephelometer, and a reduced range (1 to 5 + amu) mass spectrometer to an expanded complement that includes a dual-channel IR radiometer, a color filtered nephelometer, additional GC/MS columns, a microphone, and an RF lightning detector. Table II-6 summarizes the various payloads.

The minimum payload was defined for a weight-limited dual-planet mission or a multiple probe mission to Jupiter. It would determine the H/He ratio, the mean molecular mass, the pressure-temperature structure, the cloud structure, and provide some indication of the cloud composition down through the 15 to 20 atmosphere pressure levels. Its measurements are independent of solar lighting conditions and can be used on either the dark or light sides. Figure II-5 illustrates a typical measurement profile and targeting.

Table II-5 Possible Instrumental Techniques for Observables

Observable	Instrumental Techniques
1. H/He Ratio	a. Palladium Column GC/1 to 4 + $\geq$ 5 m/e MS
	b. 1 to 4 + $\geq$ 5 m/e MS
	c. Nominal GC/MS
	d. Expanded GC/MS
2. Isotopic Ratios	a. Nominal GC/MS
	b. Expanded GC/MS
	c. H/D Photometer (4.55 μ CH_3D)
3. Mean Molecular Mass	2.a. or 2.b. Above/Possibly 1.b.
4. Minor Constituents	2.a. or 2.b. Above
	CH_4, NH_3 Absorption Photometers
5. Temperature/Pressure	a. Pressure Gages of Various Types
	b. Immersion Thermometers (Thermocouples, Pt Wire)
	c. 10 μ Radiometer for Relative Temperature to ± 0.1°K
6. Cloud Layering	a. Nephelometer
	b. Aerosol Photometer (Cloud Tops Only)
	c. IR Radiometers (5 μ and 10 μ)
	d. Pressure/Temperature - Very Precise
7. Cloud Composition	a. Nominal or Expanded GC/MS
	b. Pressure, Temperature, Nephelometer
8. Cloud Colors	a. Nephelometer with Color Wheel
	b. Photometers (Solar) with Color Wheel
9. Solar Flux	Photometers with Various Filter
10. IR Flux	Up and Down-Looking IR Photometers (Various Wavelength Bands)
11. Complex Molecules	a. Nominal or Expanded GC/MS
	b. UV Spectrophotometer with Light Source
12. Lightning	a. RF Lightning Detector and Microphone
	b. Optical Flash Detectors
13. Cloud Particle Sizes, etc	a. Cloud Particle Counter
	b. Nephelometer
14. Turbulence	a. Accelerometers
	b. Pressure/Temperature Fluctuations
15. Magnetic Fields	Various Magnetometers
16. Electric Conductivity	a. Electrometer
	b. Loss of RF Communications Link
17. Li, Be, B Ratios	Nominal or Expanded GC/MS
18. Ionosphere Composition	a. Ion Mass Spectrometer
	b. Neutral Mass Spectrometer
19. Upper Atm Temperature	Electron Density Probe and Ion and Neutral Scale Heights
20. Decameter Radiation	Receivers on Probe and Spacecraft

Table II-6 Summary of Science Payloads

M[illegible] (Mission D) (~ 18 atm)	Nominal (Parametric and Design Example) (50 to 1000 atm)	Expanded (Mission B) (75 atm)
1 to 5 am Mass Spec	GC/MS	Expanded GC/MS
Pressure, Temperature	Pressure, Temperature	Pressure, Temperature
Nephelometer	Photometers	Nephelometer (Color Filters)
	Accelerometers (Entry and Turbulence)	Visual Photometers
		IR Radiometer (5μ and 10μ)
		RF Lightning Detector and Microphone
		Accelerometers (Entry and Turbulence)
(10 lb 11 bps)	(16 to 25 lb 33 bps)	(27 lb 30 bps)

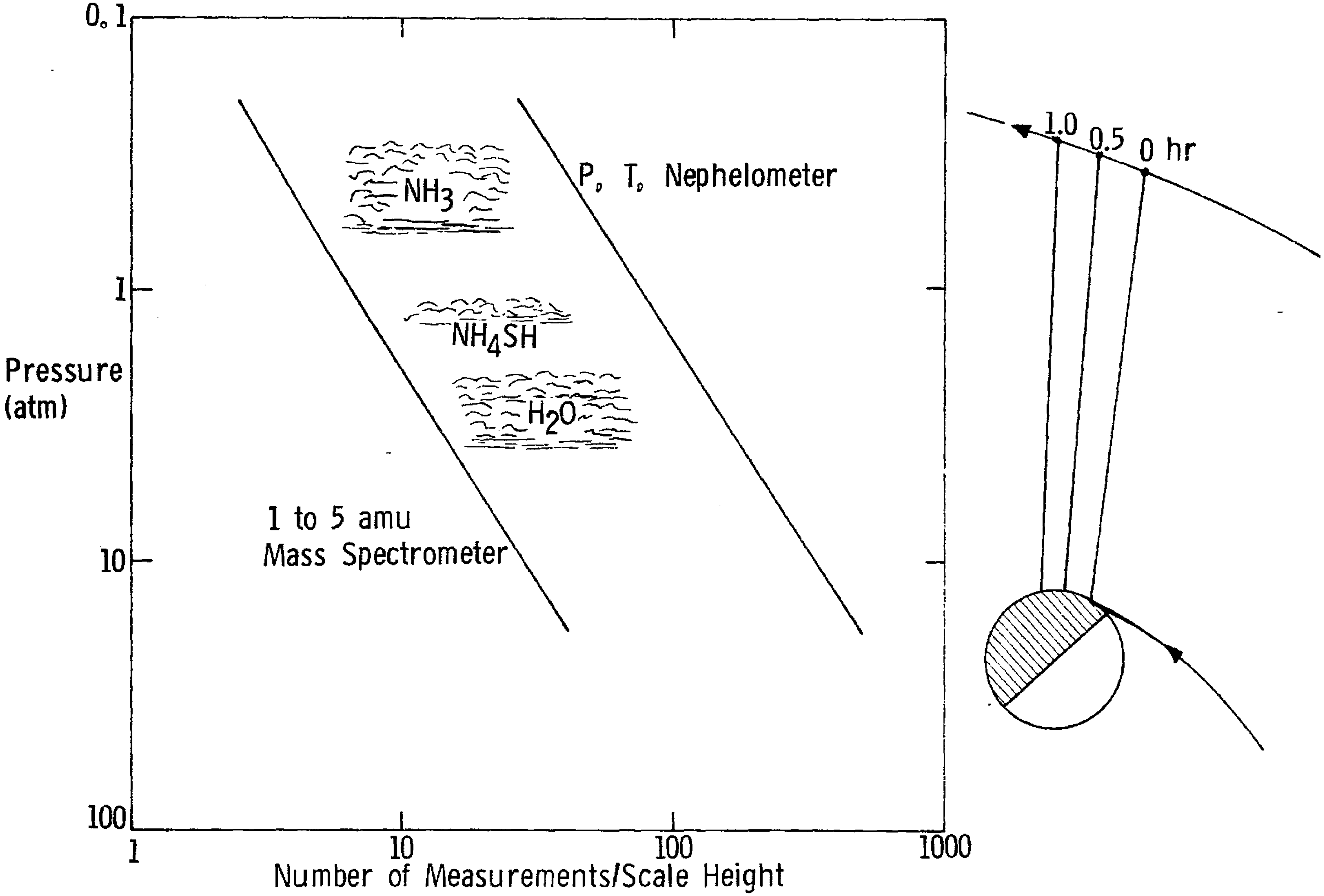


Fig. II-5 Typical Minimum Payload Measurement Profile

The nominal payload was specified by JPL for use as a baseline. Its targeting and measurement capabilities are illustrated in Fig. II-6 and II-7. An ion mass spectrometer is also included for determining the ion concentration profile from 3 Jupiter radii down to the point where blackout occurs.

The expanded payload was defined to improve the science accomplishment over that of the nominal payload. Figure II-8 illustrates a typical measurement profile and targeting for the expanded payload. While the mission illustrated reaches a depth of only 75 atm, its effectiveness for providing scientific information relevant to the observables is about 30% greater than a nominal payload carried to 300 atm. This is due to the more optimum instrument complement. On the other hand, the minimum payload carried to a depth of 18 atmospheres is about 50% less effective than the nominal payload to 300 atmospheres. A comparison of the scientific effectiveness of the various missions studied is given in Volume II, Chapter IV.

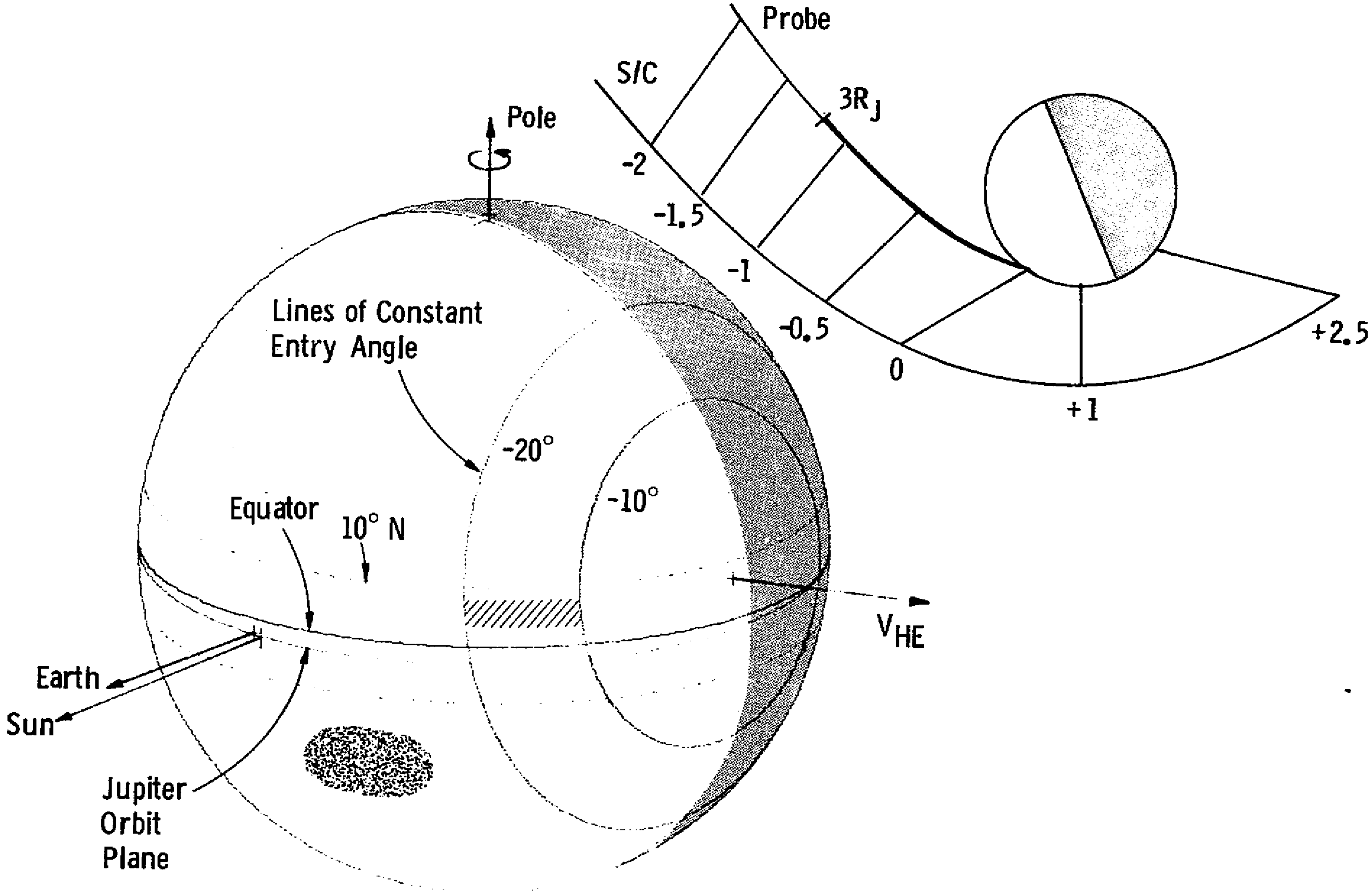


Fig. II-6 Nominal Payload Targeting

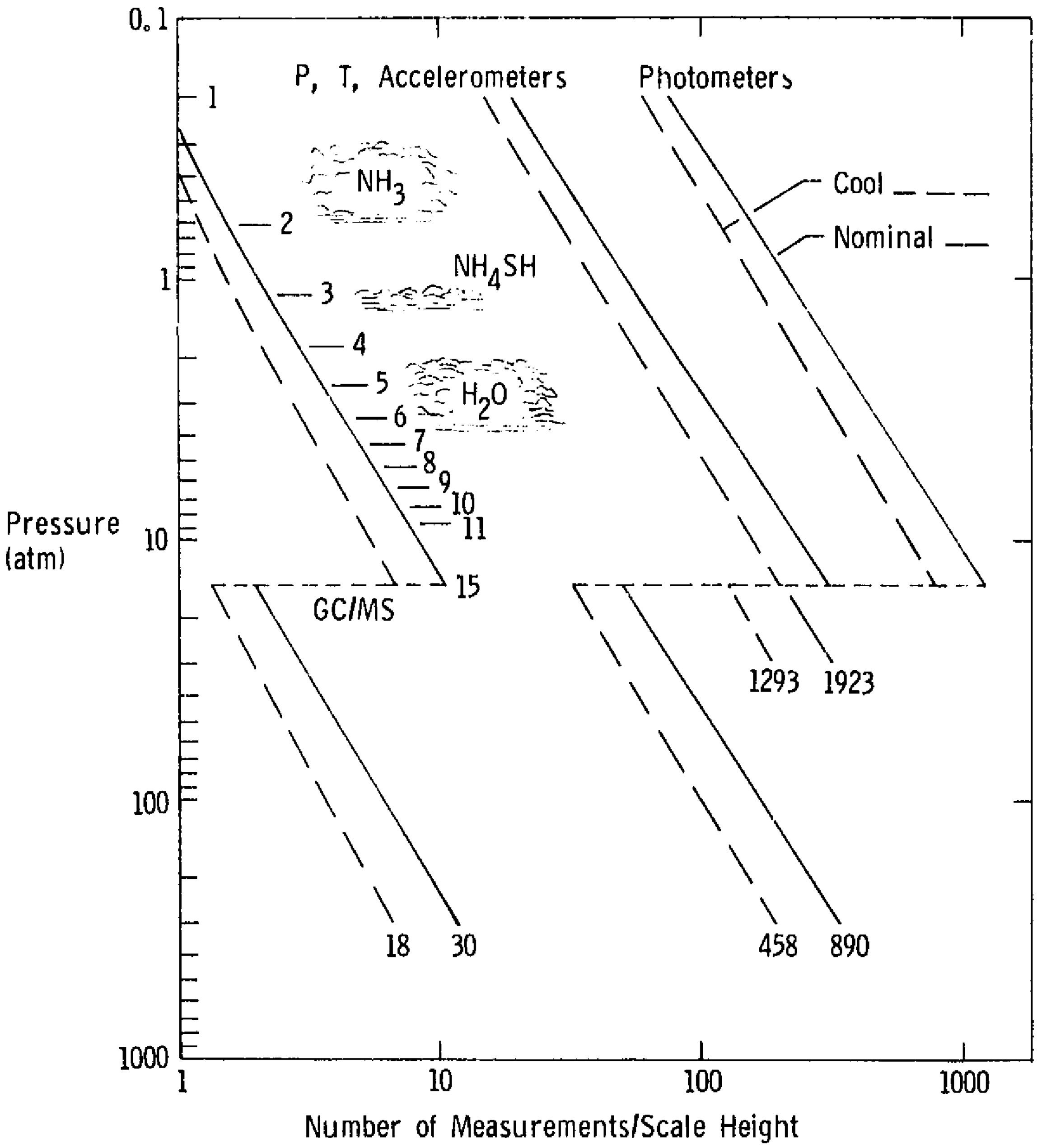


Fig. II-7 Measurement Profile for Nominal Payload

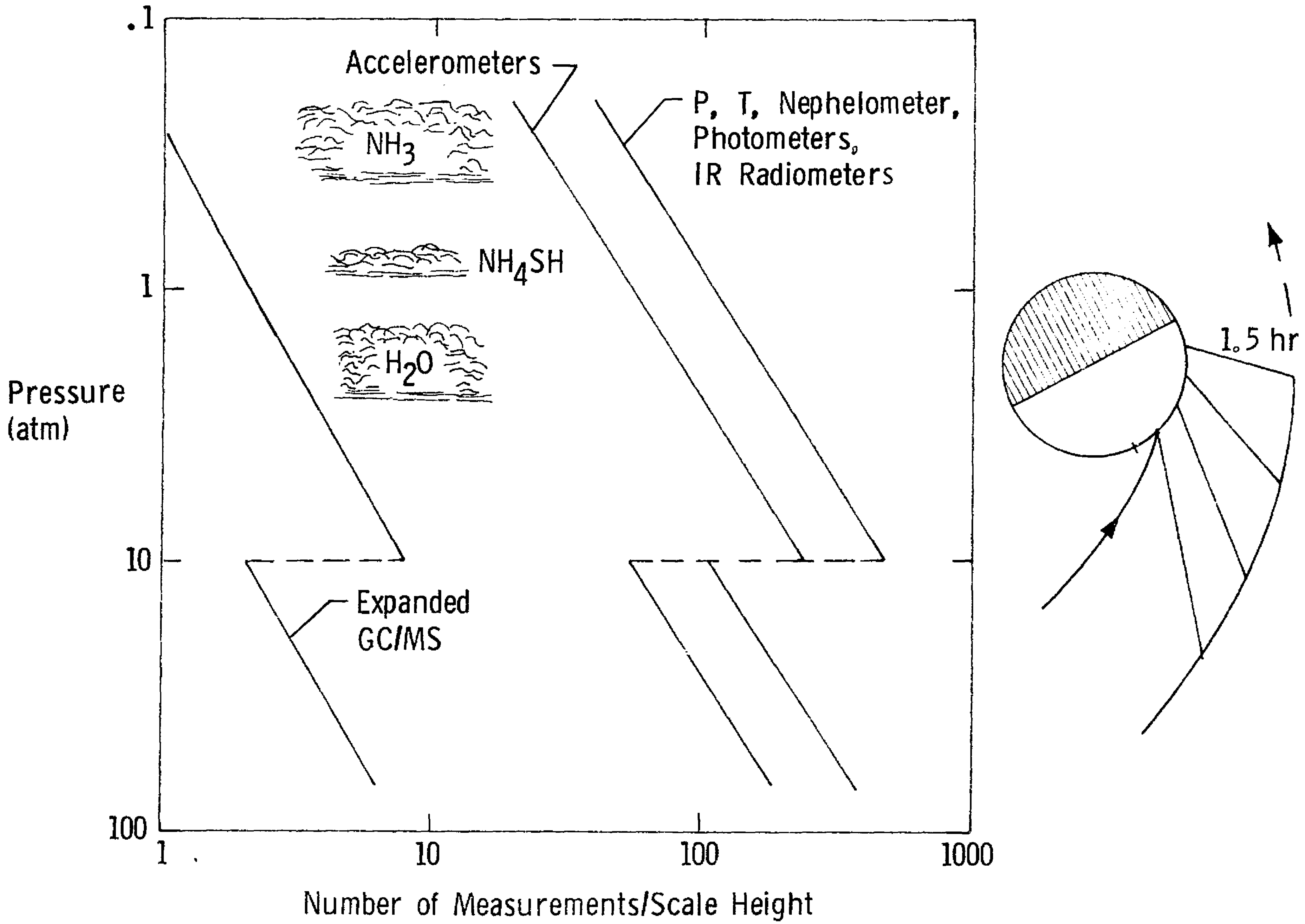


Fig. II-8 Typical Expanded Payload Measurement Profile

C. MISSION SELECTION

The mission selection approach, with many variations and constant iteration, consists of choosing the science mission objectives, appropriate science payload, and depth required, and working backwards through the mission profile with the descent, entry, and trajectory factors to establish the total launch weight required. This can then be compared with the available launch vehicle capabilities and a launch vehicle can be selected.

Figure II-9, Mission Selection Factors, identifies the major trade considerations associated with the phases of the mission profile.

For this study, the science objectives were specified in the ground rules. From these rules a set of science questions were developed and the relevant measurements and measurement intervals for each of the instruments were determined. Although a nominal payload was specified in the ground rules, both reduced and expanded science payloads were defined. The nominal payload, consisting of a gas chromatograph/mass spectrometer, photometers, and an aerometry experiment with pressure, temperature, and accelerometers, was considered the minimum payload that could answer the more complex questions involved with composition. However, the reduced, or minimum payload is still considered adequate for resolving the major questions, although it has a reduced capability mass spectrometer with no gas chromatograph, a nephelometer to replace the photometers, and accelerometer data are taken only during entry. Pressure and temperature measurement capability remains the same. The expanded payload adds capability in the gas chromatograph/mass spectrometer, and adds a nephelometer, IR radiometer, and a lightning detector.

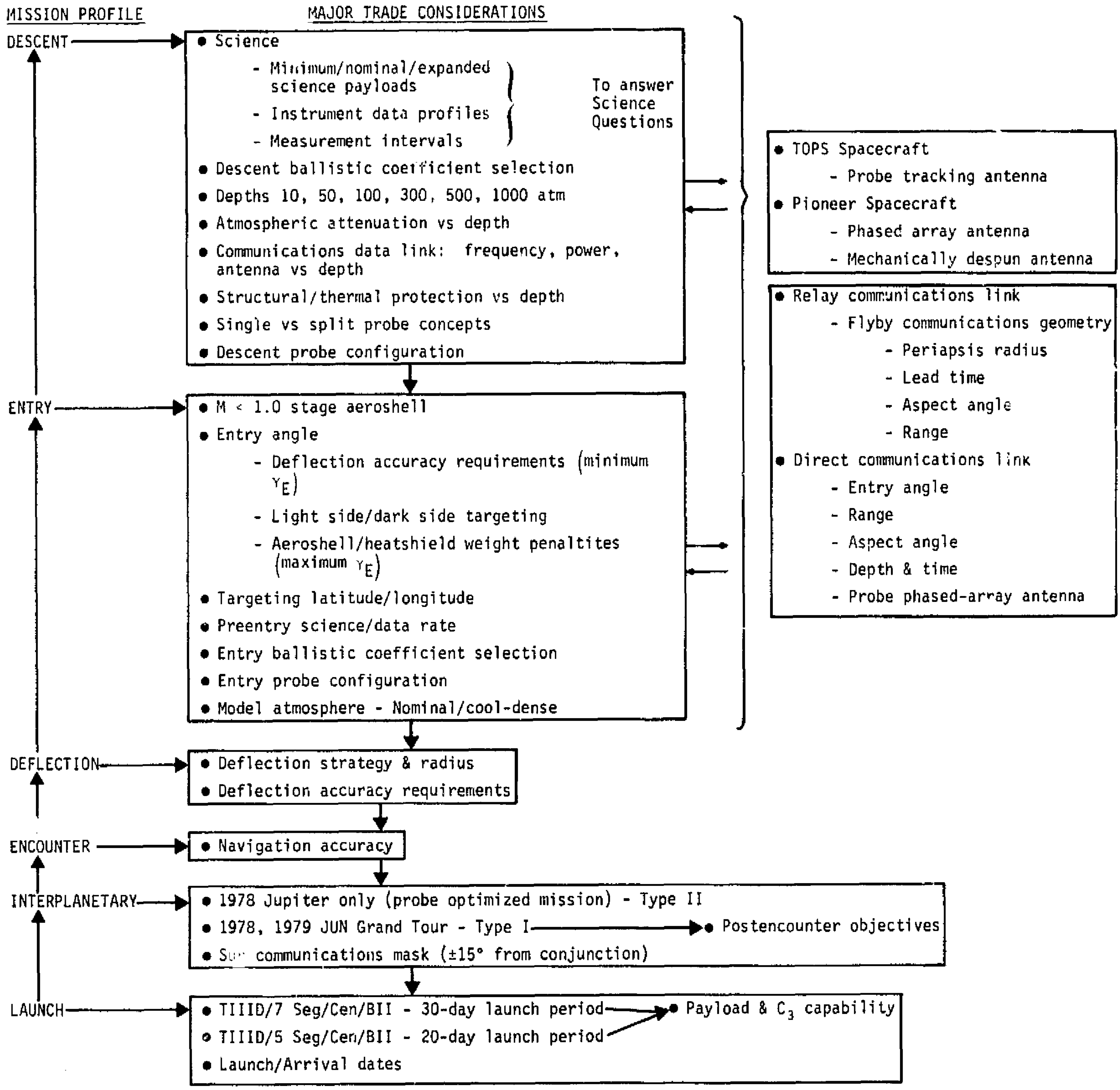


Fig. II-9 Mission Selection Factors

Referring to Fig. II-9 the science measurement intervals and data rates determine the desired descent profiles to various depths and from these, the probe ballistic coefficients are selected that provide the best match. The communications link (to the flyby spacecraft or a direct link to earth) is sized as a function of depth along with the structural/thermal protection system. The integration of these subsystems results in the descent probe configuration. For depths greater than pressures of about 300 atm, the greater descent times and the increasing atmospheric attenuation result in such severe transmitter power and weight penalties, that a split probe concept was required. This concept includes a lower probe transmitting to an upper probe relay, which in turn transmits to the relay spacecraft. Figure II-10 shows a weight comparison of the single and split probe concepts versus depth. The split probe has a considerable weight advantage at the greater depths, at the cost of added complexity.

As indicated in Fig. II-9 the descent probe design, and especially the data link, must be integrated with the choice of spacecraft, the required flyby geometry, and the resultant antenna pointing angles. This critical spacecraft/probe communications geometry interface was one of the major system integration problems of the study. When optimized for the descent probe mission, a spacecraft flyby periapsis of about 2 Jupiter radii results. Figure II-6 illustrates the synchronization of the probe's position in the planet's rotating atmosphere with the flyby spacecraft. This flyby radius of about 2 R_J minimizes the combined losses due to probe aspect angle, range, and atmospheric attenuation, for a 2½ hr mission. When the entry mission is tied into a Grand Tour mission, where the flyby radius is determined by the postencounter objectives, some degradation in the probe mission results.

The descent probe configuration and weights are then factored into the entry vehicle design along with entry angle requirements, heat shield/aeroshell weight factors, and ballistic coefficient values required to meet the subsonic staging of the aeroshell at about 0.2 atm pressure, Fig. II-11. Although shallow angles are desirable from a weight standpoint, the minimum entry angle is limited to about -10 deg due to deflection accuracy tolerances, and the maximum entry angle range is constrained by system weight to angles below about -50 deg. Figure II-12 shows the sensitivity of this trade. Within the usable entry angle range, the entry angle is chosen based on targeting and lightside/darkside considerations. For 1978 Type II launches, entry angles between -7 and -20 deg are required for 1 hr of daylight descent time. The corresponding entry angles for Type I trajectories are between -24 and -48 deg.

From the above trades the entry probe can be defined and the deflection strategy, radius, ΔV and application angle are determined.

The probe system is now integrated with the spacecraft (TOPS or Pioneer) and the total weights can be matched with available missions and launch vehicle capabilities. The interplanetary trajectory presents one principal tradeoff; the differences in Type I and Type II paths. For example, payload capability, and the desire for daylight entry at Jupiter favor Type II paths. Mission duration favors Type I paths. The application to post-encounter missions such as the Grand Tour requires Type I paths.

Early study results showed that there are many feasible missions. Therefore, to illustrate a typical cross section, six sample missions, A through F, were chosen, each to examine a particular characteristic of interest. In addition, a parametric trade was done to show the effect of mission depth and other parameters on the probe system design.

Fig. II-10 Effect of Depth on Descent Probe Weight

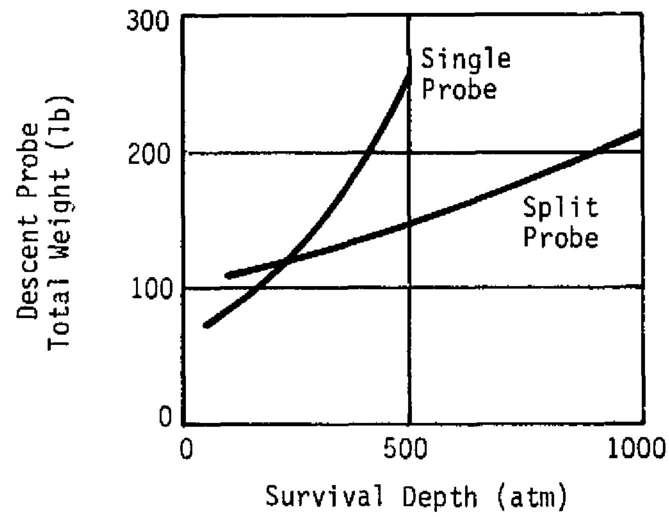


Fig. II-11 Entry Ballistic Coefficient Required for Subsonic Staging at P = 0.2 atm

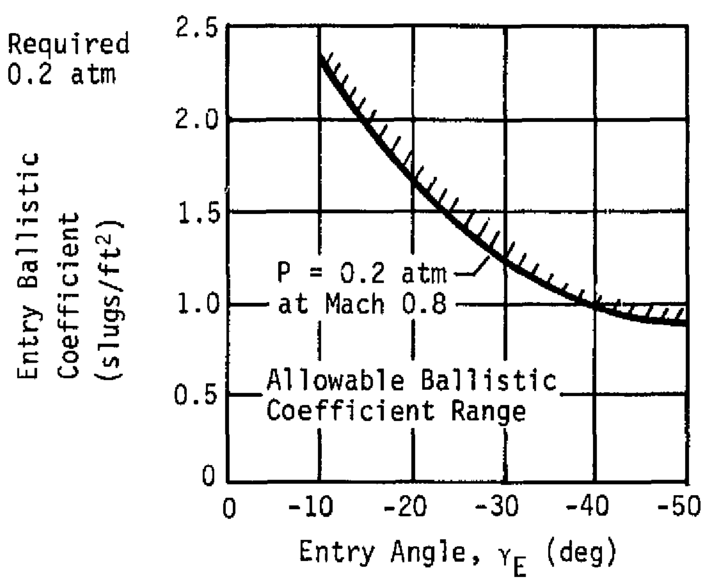
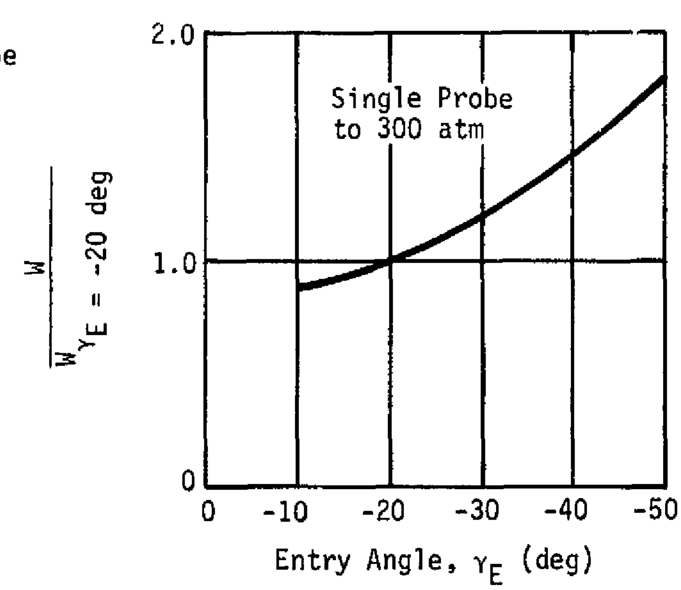


Fig. II-12 Effect of Entry Angle on Probe System Weight



The parametric trade, discussed in Section D of this chapter, examines the effects of depths to 1000 atm, entry angle, payload variation, bit rate, descent probe concept, and spacecraft flyby radius on the system weight. From this parametric study, one design was singled out and identified as the design example to illustrate detailed subsystem designs, and to examine in some depth the interface with the TOPS spacecraft. This mission carries the nominal science payload to the 300 atm depth and is delivered from a TOPS spacecraft, flying a 1978 Type II trajectory with an optimized flyby radius of 2 R_J. With a 40 bps data rate and a 40 watt transmitter, this total probe system weighs 560 lb and can be launched with a 7-segment Titan IIID/Centaur/Burner II launch vehicle. The design example is discussed in Section E of this chapter.

D. IMPLEMENTATION TRADEOFF STUDY PARAMETRIC RESULTS

To determine the feasibility of penetrating to various depths or pressure levels in the atmosphere, and to determine the subsystem design penalties incurred with increasing depth, a parametric study of probe system weight and power requirements was performed. The relay data-return mode was selected as the basis for this study because of the higher performance it afforded. The spacecraft radius of periapsis determined from the trial mission study to be most favorable to the relay link geometry, $R_P = 2.0\ R_J$, formed the baseline, but larger values, up to $R_P = 6.8\ R_J$, also were examined. Probe entry angles from -10 to -50 deg were considered to cover conditions required in achieving lightside missions.

A list of parameter variations is given below:

Science Payload	= 8 lb, 16 lb, 32 lb;
Survival Depth	= 50, 100, 300, 500, 1000 atm;
Atm Model	= Nominal and cool-dense;
Entry Angle	= -10, -20, -30, -50 deg;
Bit Rate	= 40/10 80/20 bps (initial/final);
Periapsis Radius, R_p	= 2.0, 4.0, 6.8 Jupiter radii;
Spacecraft	= TOPS and Pioneer.

The probe configurations were tailored to each of the five survival depths examined, and included both single probes and dual-relay or split probes. The upper part of the split probe collects data itself and also relays data to the flyby spacecraft from the separately descending lower part of the probe, see Fig. II-13. The basic configurations and other characteristics of the two probe types used, shown typically in Fig. II-14 and II-15, were evolved through tradeoffs discussed in Volume III. Other ground rules, assumptions, and definitions of significance in the parametric study are the following:

1) 1978 launch opportunity;
2) The nominal science payload consists of 16 lb in the descent phase including: temperature, pressure, accelerometer, gas chromatograph/neutral mass spectrometer, and photometers. A 3-lb ion mass spectrometer is contained in the entry vehicle for preentry measurements;
3) A single probe is defined as a probe whose postentry descent profile is controlled by making no more than one stepwise change in the ballistic coefficient after the aeroshell separation;

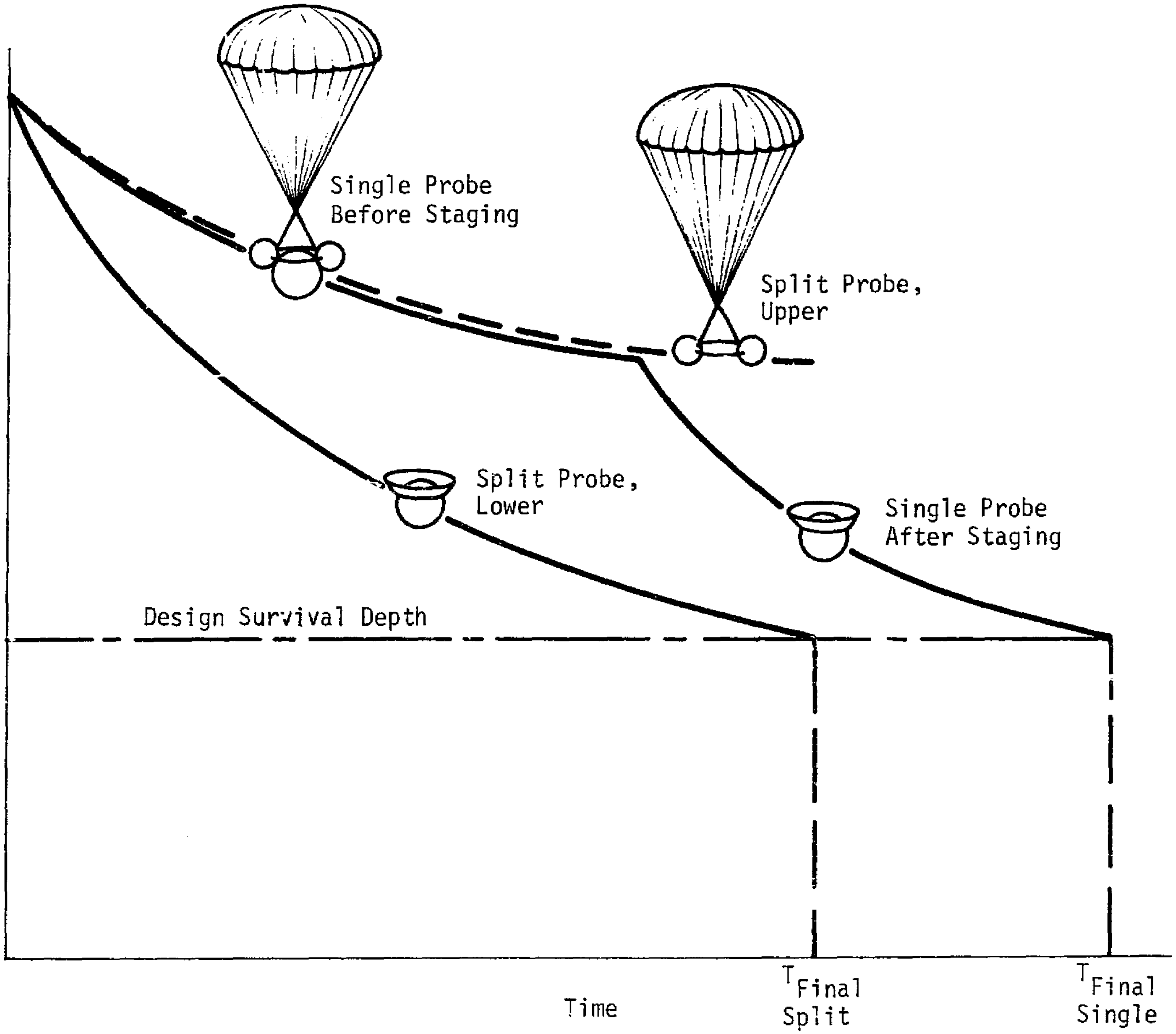
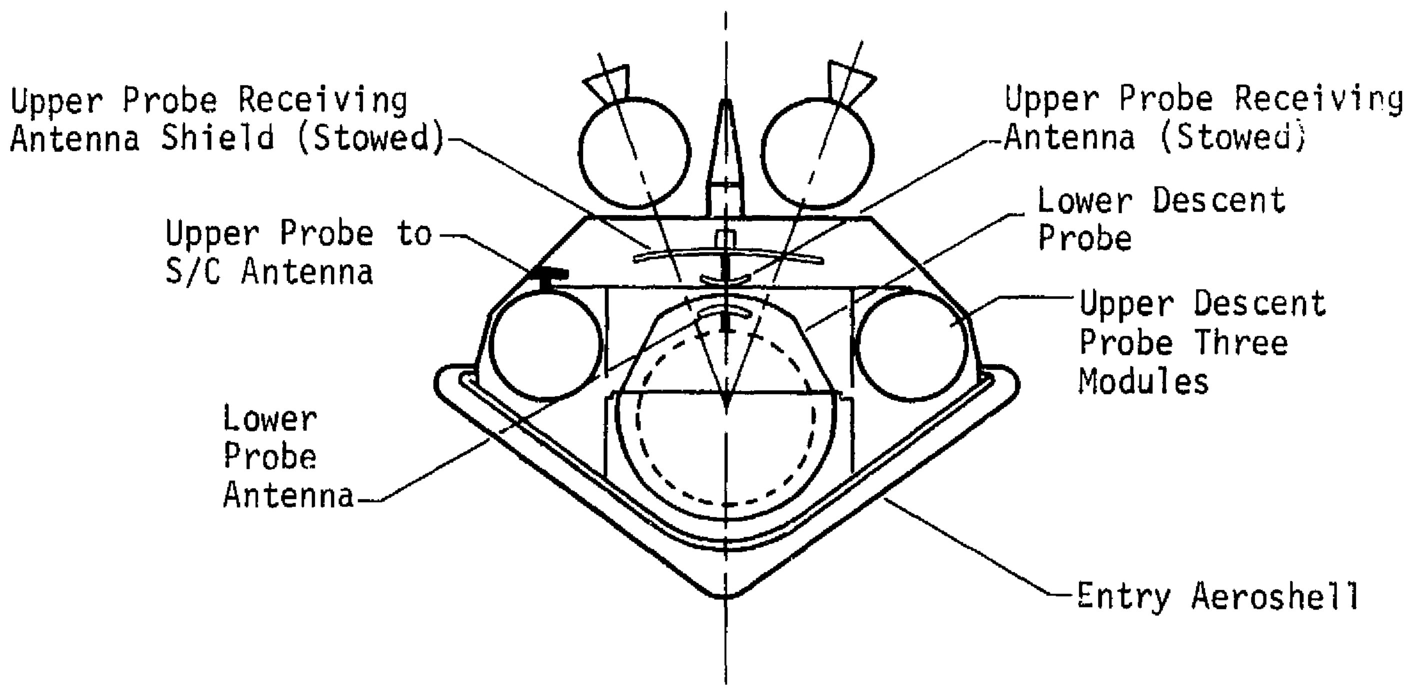
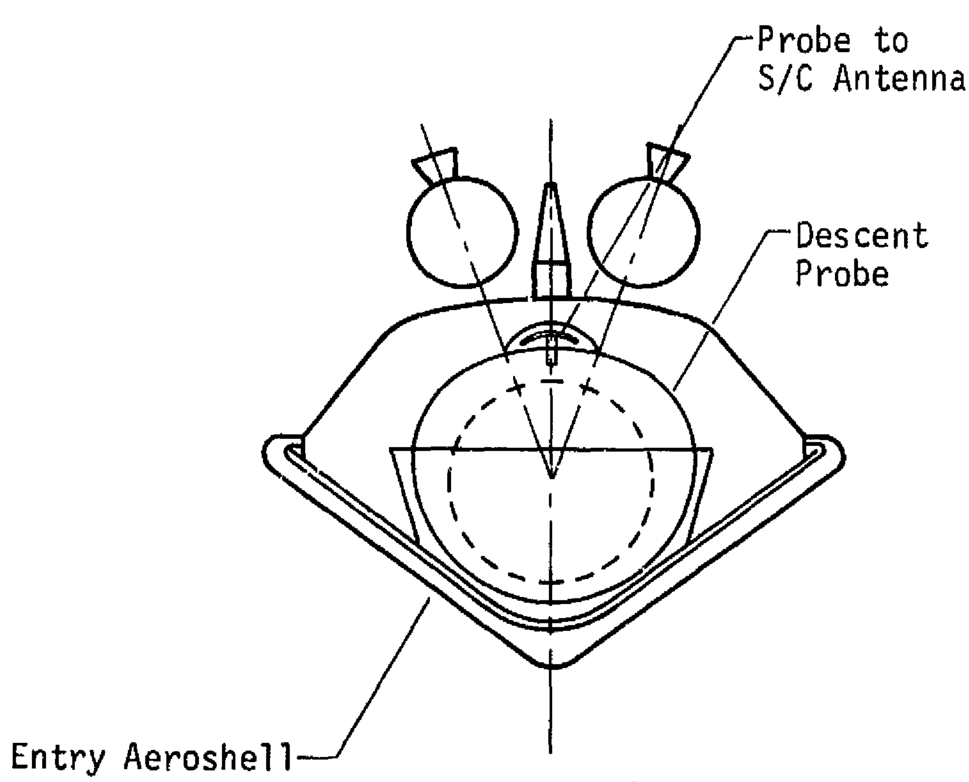


Fig. II-13 Descent Profiles for Single and Split Probes



Split Probe



Single Probe

Fig. II-14 Single and Split Probe Configurations

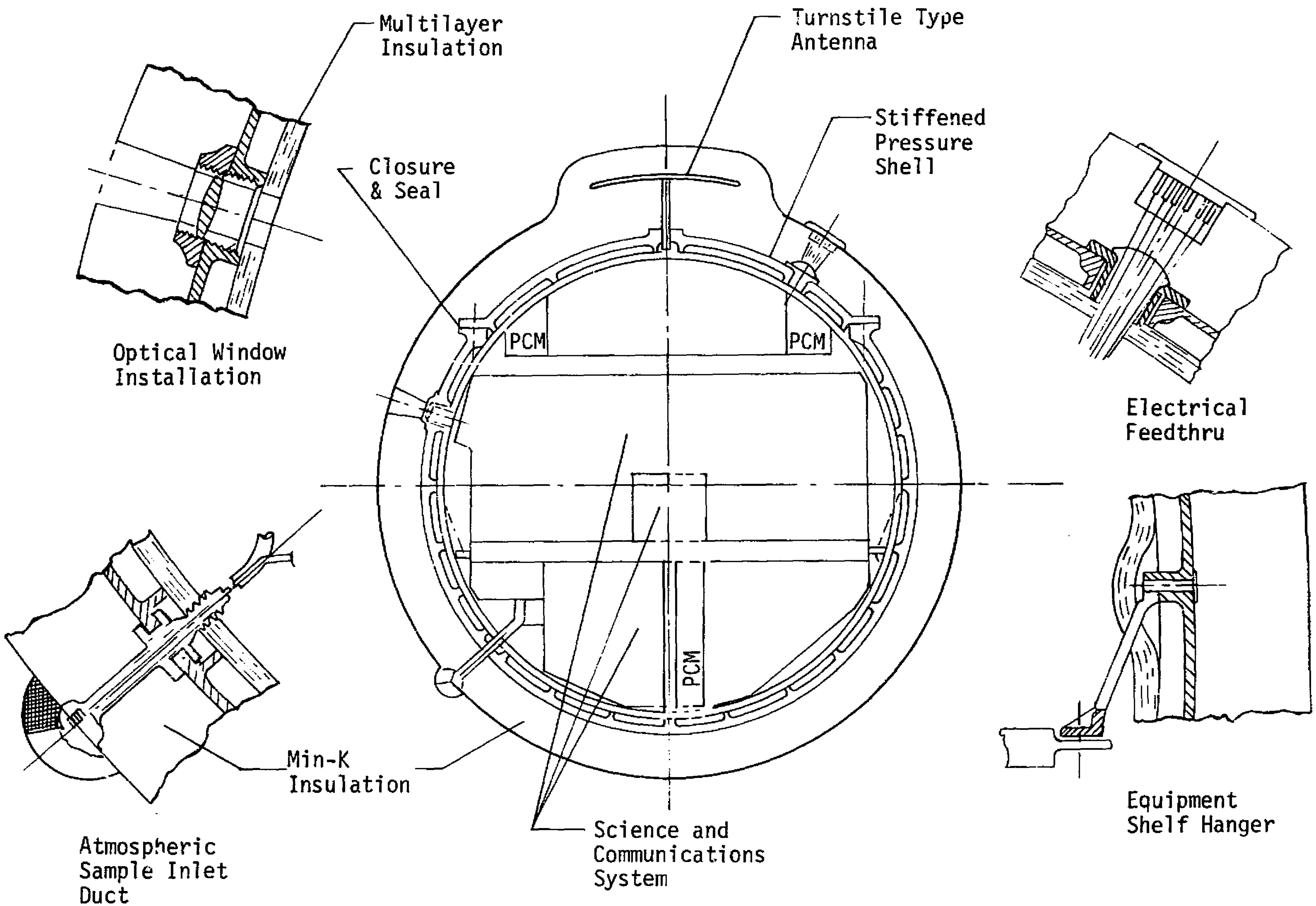


Fig. II-15 Descent Probe Structural/Thermal Concept

4) A split probe is defined as a probe whose upper and lower parts are separated at the time of aeroshell separation and then descend simultaneously;

5) Common science instrument sampling rate criteria, considered to be the minimum which achieves the science objectives, are used for all descent profiles. (Two samples/scale height for GCMS; 50 to 100 samples/scale height for temperature and pressure.);

6) Probe ejection range = 2.6 x 10^7 km. Selected on basis of being large enough to yield moderate probe ΔVs ($\sim$80 m/s) and small enough to permit passive thermal control after separation. It was also chosen with regard to avoiding occultation of spacecraft by the sun at time of probe ejection.

7) A 5-ft-diameter dish was assumed the maximum size for the probe autotracking antenna.

1. Probe Implementation Procedure

For any combination of survival depth, payload, and entry conditions the probe implementation is accomplished in the steps outlined in Fig. II-16. The tradeoffs involved are also indicated in that figure. In the final step, the total probe system weight required is obtained. It should be noted that the determination of the probe descent profile, Item 2, fixes a unique time-temperature-range relation for each survival depth. This relationship is implied in all the conclusions concerning probe performance and achievable survival depths. Use of the two-stage ballistic coefficient profiles to achieve the sampling criteria is arbitrary, but appears to be a reasonable compromise based on considerations of the increasing complexity with the number of stages vs the shortening of descent time.

1. **ESTABLISH SCIENCE INSTRUMENT SAMPLING RATE**

   Establish requirement samples per scale height & sample processing time for each instrument

2. **DETERMINE MINIMUM DURATION DESCENT PROFILE**

   Calculate 2-step profile which yields minimum descent time consistent with 1

3. **DETERMINE OPTIMUM COMMUNICATIONS FREQUENCY**

   Trade atmospheric attenuation against system temperature to minimize link losses

4. **ESTABLISH OPTIMUM RELATIVE SPACECRAFT PROBE GEOMETRY**

   Select lead time to balance & minimize probe transmitter power. Trade range, aspect angle, & atm attenuation

5. **DEFINE DATA AND TELECOMMUNICATIONS PACKAGE REQUIRED FOR DESCENT PROBE**

   Size the data handling, memory battery systems, etc, for the given science payload and descent time

6. **ESTABLISH DESCENT PROBE THERMAL/STRUCT/AERODYNAMIC DESIGN & WEIGHT**

   Conduct tradeoffs to determine optimum pressure vessel structure-insulation combination. Configure to give required m/C_DA by addition of aerodynamic features, fences, skirts, etc

7. **ESTABLISH ENTRY AEROSHELL STRUCTURE & HEAT SHIELD DESIGN; DETERMINE ENTRY WEIGHT**

   Size A/S to house descent probe; check that entry m/C_DA meets altitude criteria for achieving subsonic velocity & config adequate for static & dynamic stability

8. **ESTABLISH PROBE DEFLECTION MANEUVER SYSTEMS - ΔV, G&C, T/C**

   Size propulsion system for given ΔV & select guidance and control system to achieve ΔV implementation accuracies consistent with comm system

9. **DETERMINE SPACECRAFT MODIFICATIONS REQUIRED**

   Size the probe structural adapter, tracking antenna, additional power requirements, etc

10. **OBTAIN TOTAL PROBE SYSTEM WEIGHT BY SUMMING ENTRY, DEFLECTION, AND S/C MODIFICATION WEIGHTS**

    Combine 7, 8, & 9

Fig. II-16 Flow Chart for Parametric Probe Implementation Study

2. Results of Parametric Probe Implementation Study

An abbreviated tabulation of the system weights and probe power requirements obtained for various combinations of conditions is given in Table II-7. The last two columns in the table correspond to items 7 and 10 of Fig. II-16. From these data the total probe system weights (probe plus TOPS spacecraft modification weight) are plotted in Fig. II-17 as a function of survival depth for a specific set of conditions; $R_p = 2.0$, $\gamma = -20$ deg and the nominal science payload. Net launch vehicle capabilities are also shown -- the 1450-lb basic TOPS weight has been subtracted out of the total 1978 opportunity liftoff payload capability.

3. Survival Depth

The achievable survival depth for the single probe concept is seen (Fig. II-17) to be limited to about 300 atm by the probe power requirements. However, if sufficiently powerful, lightweight transmitters were to become available, over 500 atm depths would be possible before exceeding launch vehicle payload limits for Type II missions. Split probes are not so constrained; 1000 atm is achievable from either power or payload considerations, but these probes do introduce greater complexity. The crossover in weight between the two concepts is seen to be at about 200 atm for the $R_p = 2.0$ case plotted.

Type I trajectory payload limits result in 250 atm and 350 atm depths for single and split probes respectively.

It should be noted that although probe weights of over 450 lb (total system weights of over 550 lb) are within launch vehicle capabilities, the spacecraft modifications begin to take on more serious proportions, e.g., a change in the thrust direction of the TOPS trajectory correction system of over 15 deg is required to handle the center-of-gravity offset.

Table II-7 Ranges of Probe Weights for Survival Depths, Entry Angles, and Periapsis Radius

Periapsis Radius	Entry Angle (deg)	Survival Depths (atm) Single Probes	Survival Depths (atm) Split Probes	Transmitter Power Required (watts)	Weight of Science, Communications & Power (lb)	Total Descent Weight (lb)	Entry Weight (lb)	Total Probe System Weight (lb)
2 R♃	-10	50		10	45.9	69.4	228	320
		500		70	83.5	258.6	601	755
			300	10	68.5	130.5	337	446
			1000	20	74.7	216.5	497	642
	-20	50		10	45.9	70.7	246	341
		300		40	63.6	156.6	427	557
			100	10	67.4	115	310	416
			1000	10	70.6	210	514	654
	-30	50		10	45.9	75.0	312	422
		300		40	63.6	162.7	573	683
	-50	50		20	49.0	83.4	478	633
		300		30	58.3	158.4	805	1016
			500	20	72.9	180.3	810	1022
			1000	20	74.7	236.9	1032	1283
4 R♃	-20	50		40	56.4	89.5	298	437
			300	30	74.9	142.0	389	552
			500	40	76.9	172.8	474	649
6.8 R♃	-20	50		113	80.5	123	410	611
			300	50	82.3	152	404	606
			500	50	84.5	183.9	489	713

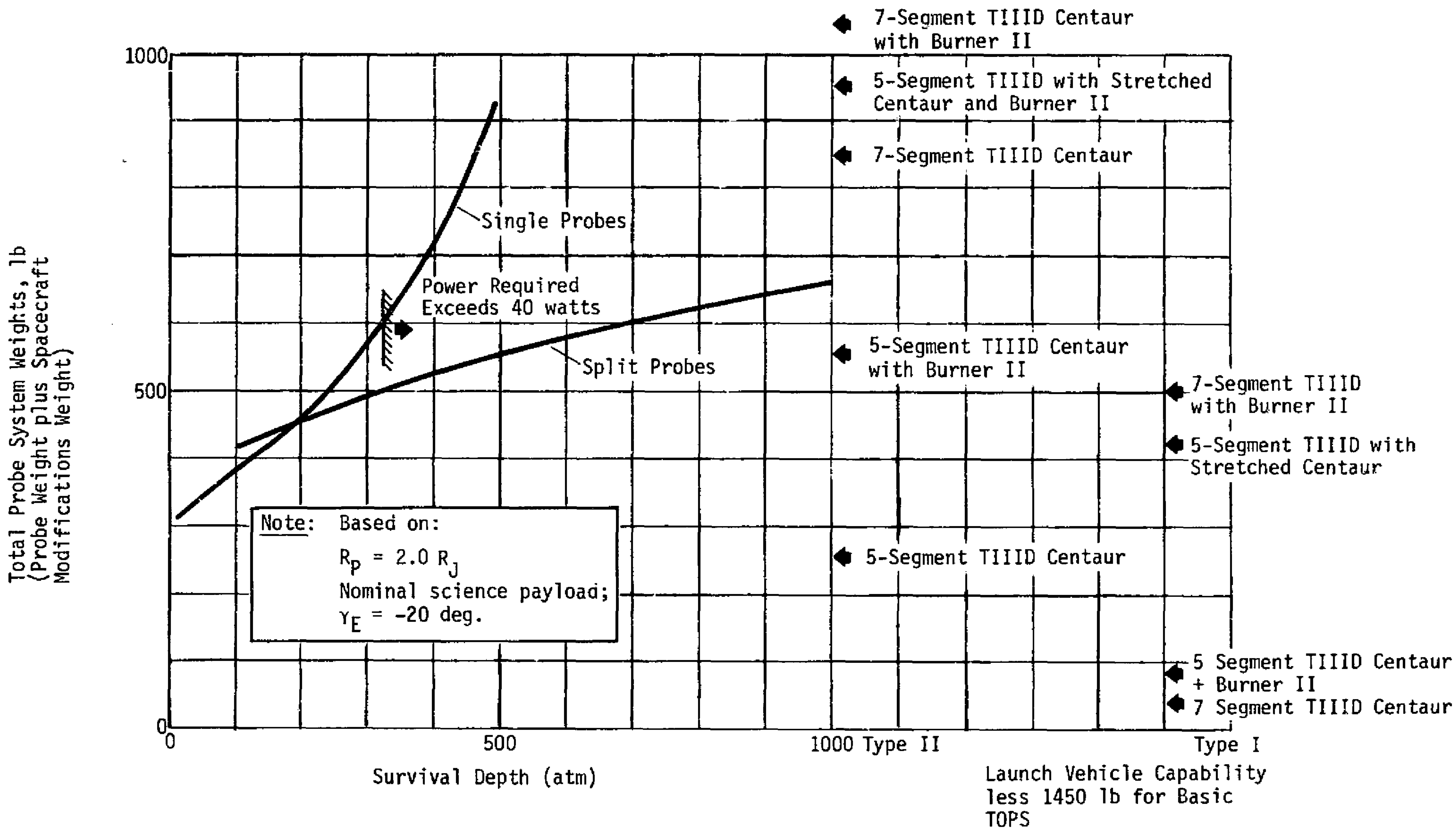


Fig. II-17 Survival Depth Achievable with Various Launch Vehicles

4. Sensitivity to Entry Angle

The effect of entry angle illustrated in Fig. II-18 is seen to be more pronounced than any factor except survival depth itself, e.g., 80% more weight is incurred at -50 deg than at -20 deg. This is primarily due to the increase in aeroshell structure weight with increasing dynamic pressure. The effect is somewhat exaggerated at the higher γ's since the aeroshell structures are optimized for the low end of the γ scale. Heat shield weight sensitivity to γ is not a contributor to the overall weight sensitivity, contrary to what might be expected. The heat shield weight fractions do not increase significantly with entry angle because the shorter entry time for steep entries offsets the higher peak heating conditions.

5. Sensitivity to Periapsis Radius

Increasing the radius of periapsis beyond 2.0 Jupiter radii is seen to result in large probe power requirements for single probes (Fig. II-19), and consequently to limit probe survival depths achievable with reasonably sized transmitters, e.g., achievable depths are less than $\sim$20 atm for R_p = 6.8. The influence on probe weight is also shown in Fig. II-19. Both power and weight sensitivities to R_p are found to be less pronounced for the split probe concept. This fact causes the single vs split probe weight crossover point, which is at 200 atm for the R_p = 2.0 case (Fig. II-17), to shift to lower pressures as R_p is increased.

Reducing R_p below 2.0 was found in the trial mission study, Volume III, to constrain probe depth by creating severe probe/spacecraft aspect angle variations, hence lower values than R_p = 2.0 were not included in the parametric implementation study.

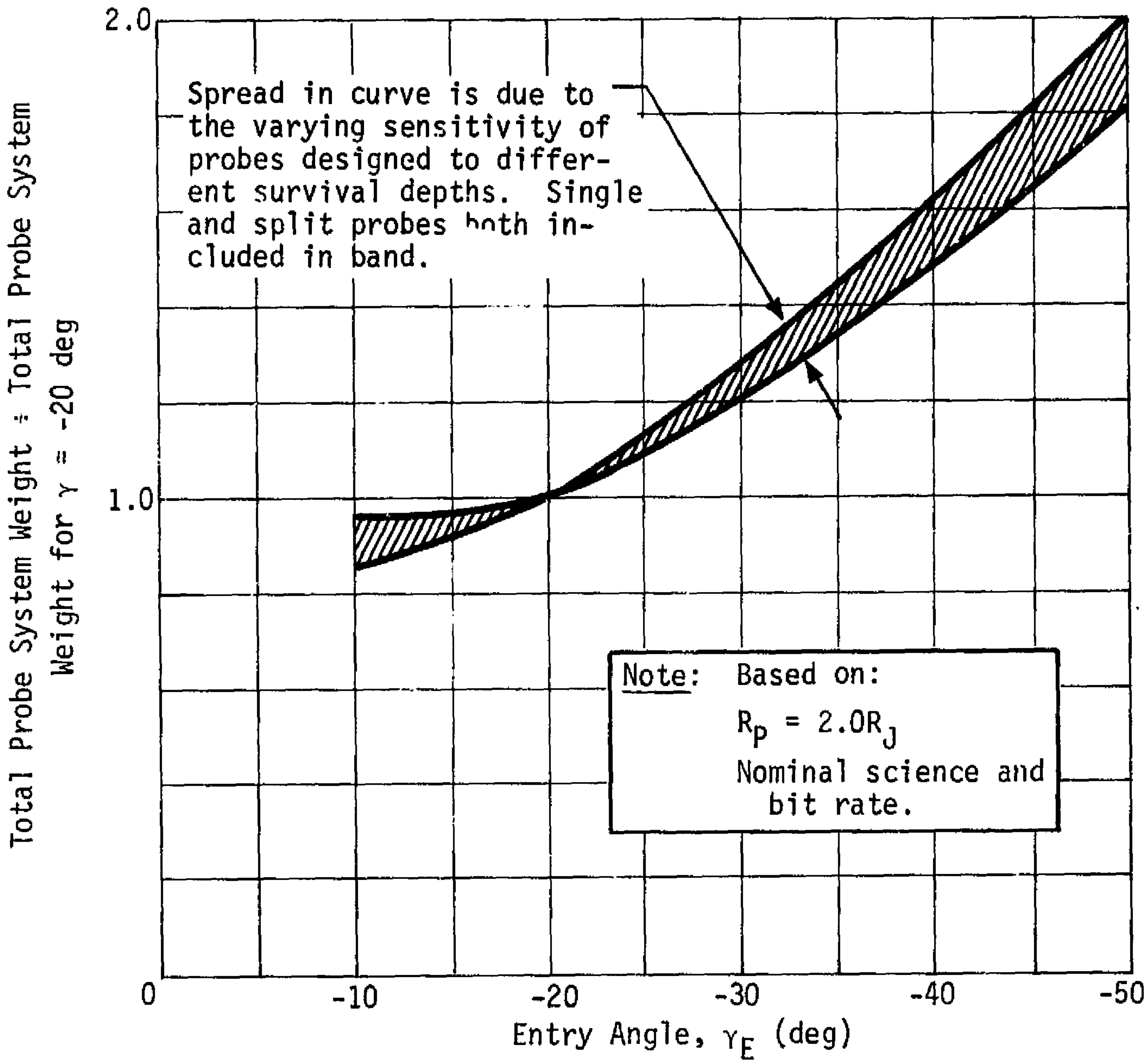
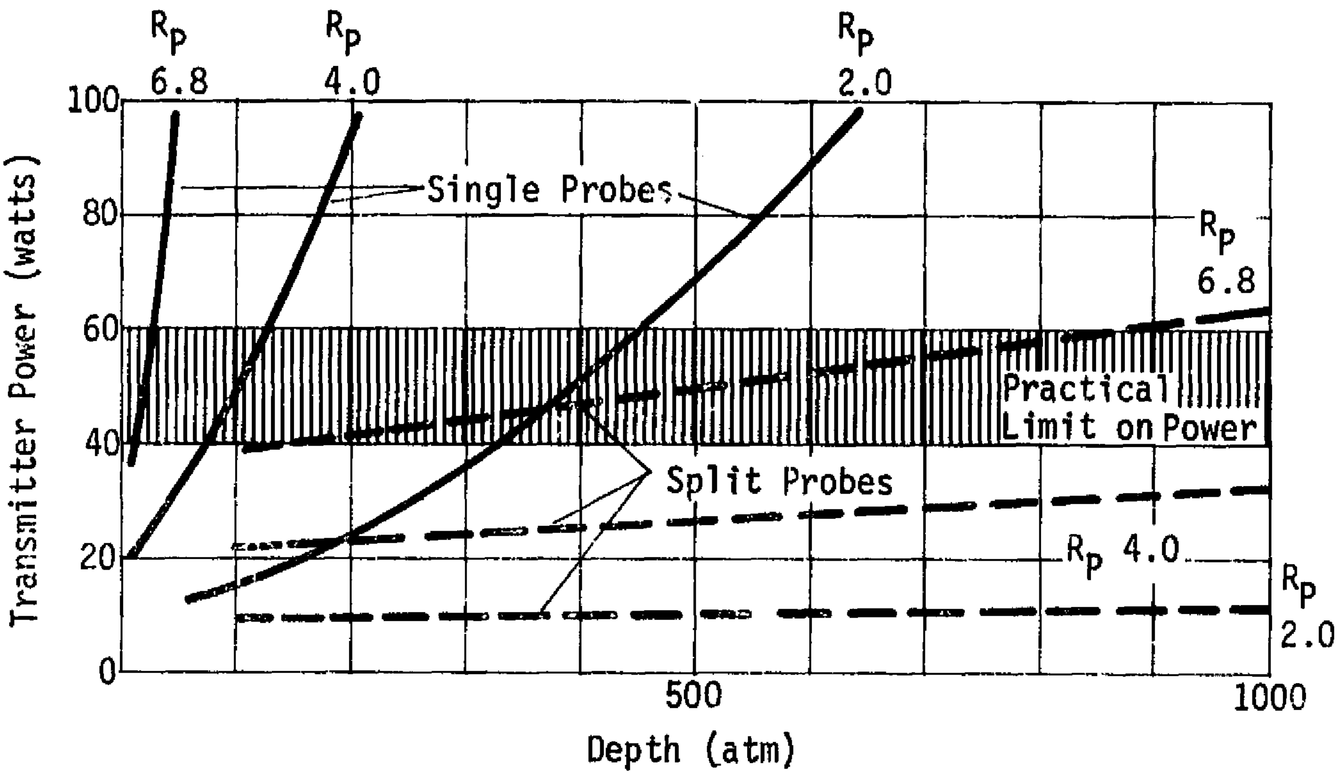
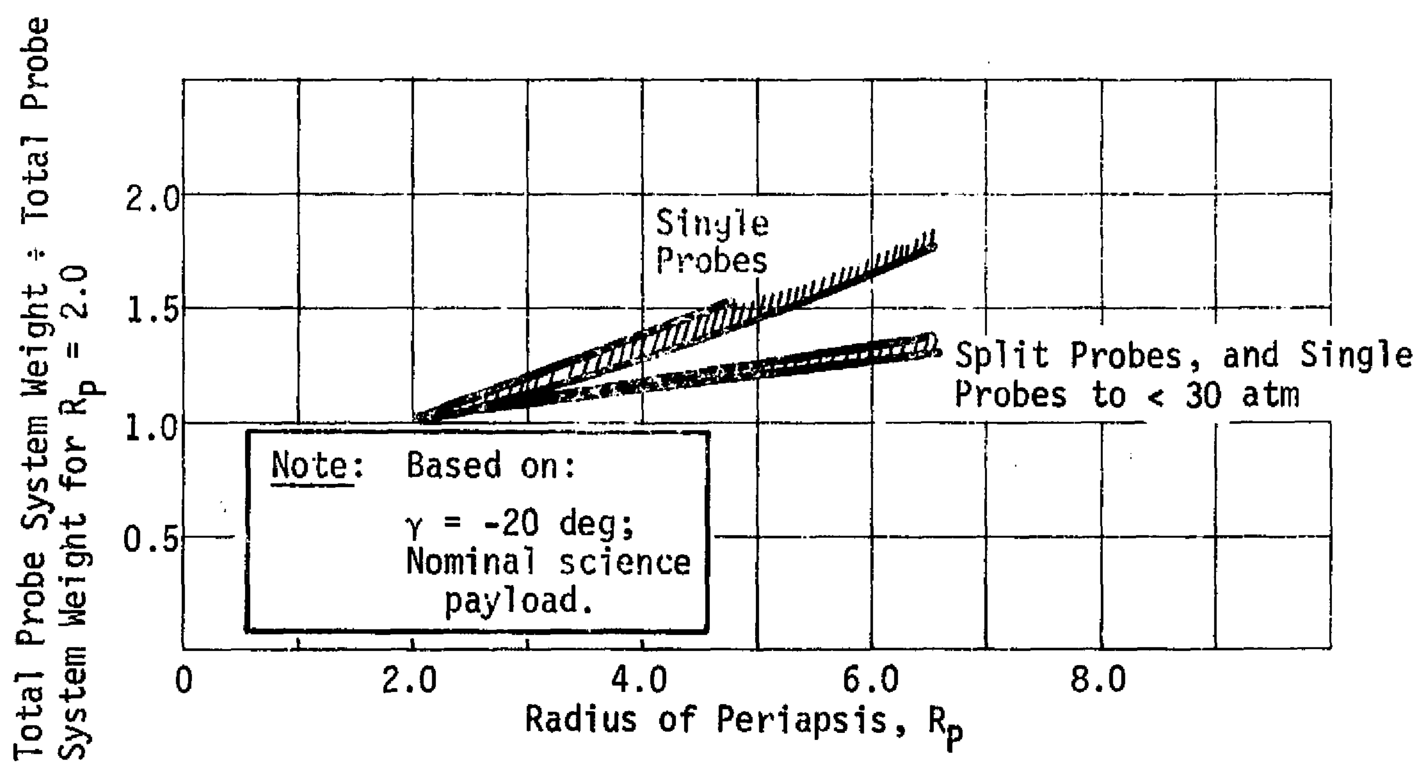


Fig. II-18 Total Probe System Weight Sensitivity to Entry Angle



(a) Power Requirement Sensitivity to R_p



(b) Weight Sensitivity to R_p

Fig. II-19 Effect of Radius of Periapsis on Power Required and Total Weight

6. Sensitivity to Science Payload and Bit Rate

The total probe-weight sensitivity to science payload is shown in Fig. II-20 and is seen to be relatively mild. The data and communications system constitutes the larger portion of the descent probe contents and since that system does not change significantly with changes in science payload, the effect of doubling the science weight results in only about a 30% increase in the total system weight. Bit rate primarily affects transmitter power requirements, and hence is more of a factor in achievable depth from that standpoint than from a weight standpoint.

7. Sensitivity to Model Atmosphere

The result of modifying a 300 atm, nominal-atmosphere-model probe to also survive to 300 atm in the cool-dense model is a 30% total system weight penalty for a probe entering at -20 deg. Steeper entries would increase this penalty for designing to both models since a part of the penalty is due to the higher dynamic pressure experienced during entry in the cool model. The other part of the penalty is due to the greater attenuation (larger transmitter required) in the descent phase of the cool atmosphere. If an unmodified nominal-atmosphere-probe encounters the cool-dense model, it will transmit data only to 30 atm (possibly to 140 atm if the bit-rate switching altitude were revised) before exceeding transmitter power requirements.

Designing a probe to go to 300 atm in *only* the cool-dense model atmosphere could be accomplished for essentially the same weight as for the nominal model probe, i.e., the reduced descent time and lower temperature work in favor of the cool model probe, but the increased atmospheric attenuation (higher transmitter power) and greater entry dynamic pressures almost exactly balance the scales. Such a cool model design would survive to only about 20 atm in the nominal atmosphere before exceeding the relatively low structural temperature to which it was designed.

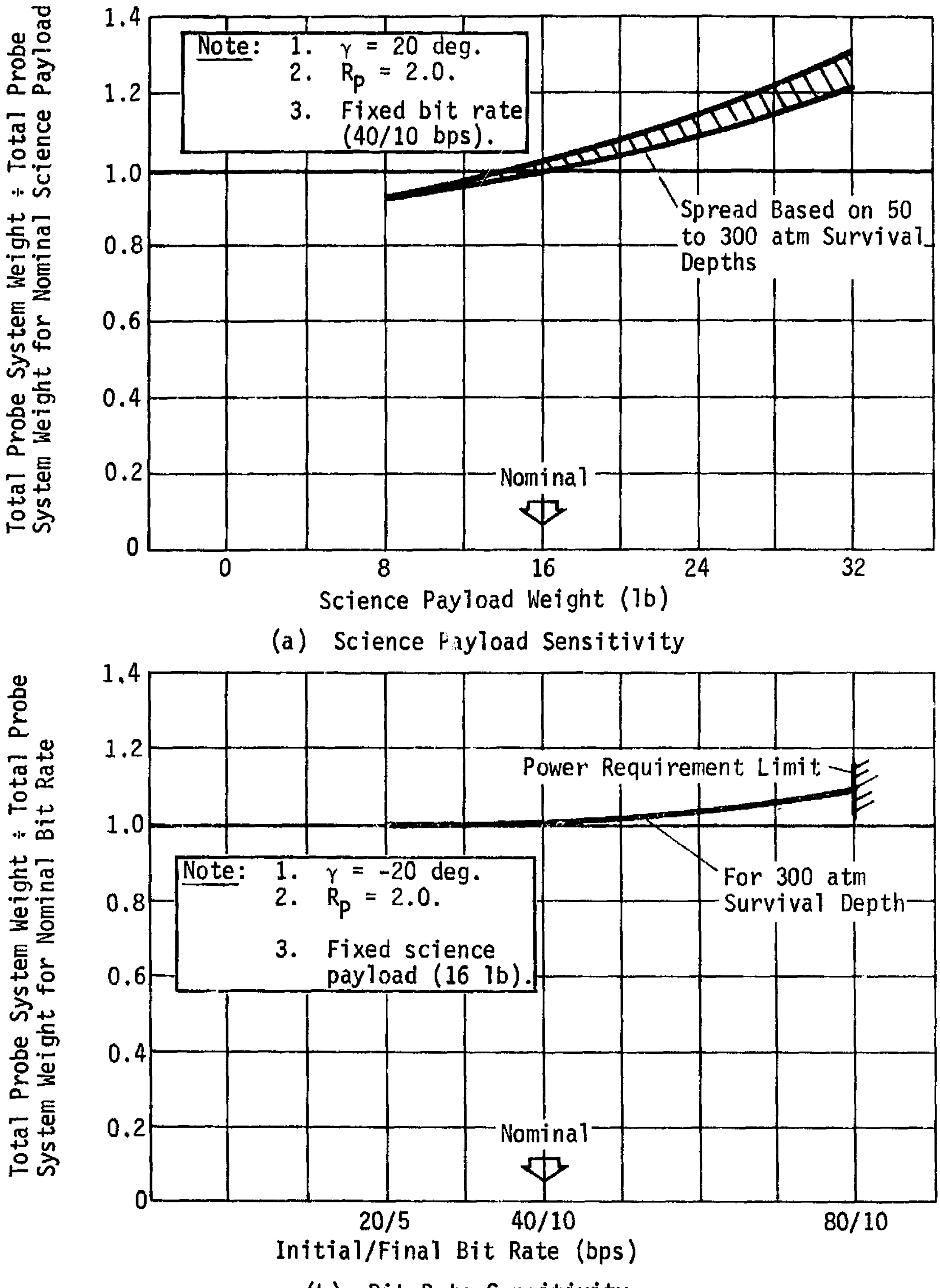


Fig. II-20 Sensitivity of Total Weight to Science Payload and Bit Rate, Single Probe Concept

The sensitivity of split probes to the extreme variation between model atmospheres is much less than that for single probes. The 1000 atm depths can be effected in both models by only changes in the telecommunications frequency.

8. Multiplanet Missions

Multiplanet missions are found to reduce the launch vehicle payload capability due to R_p constraints. Probe/spacecraft communications are also degraded by the resulting R_p values. Therefore smaller, shallower probes are required if they are to be combined with Grand Tour missions, e.g., depths of approximately 10 atm and 100 atm for single and staged probes, respectively, in the case of the 1979 Jupiter-Uranus-Neptune mission, and 100 atm for a single probe in the case of a 1978 JUN mission. These combined probe/Grand Tour missions were explored further as sample missions A & E of Section E of this chapter.

9. Use of Pioneer Spacecraft

Although the above study was performed with TOPS as the spacecraft, due to its greater compatibility with tracking the probe over a wide range of spacecraft/probe aspect angles, it is possible also to consider Pioneer as the delivery vehicle for the entire range if a mechanically despun, boom-mounted dish antenna can be implemented. Preliminary investigations indicate that this approach is feasible. However, the impact of potentially greater probe trajectory dispersions (due to the less accurate probe deflection implementation capability of Pioneer) requires further evaluation. If these areas can be satisfactorily resolved, any probe depth constraints due to payload limitations are removed, i.e., although probe system weights and spacecraft modifications weights are somewhat greater for the case of Pioneer the combined effect is only about 200 lb compared to a difference in TOPS and Pioneer basic weight of 900 lb. Probe power-requirement constraints remain, and will likely be aggravated by the increased dispersion in footprint.

The probe and antenna mounting provisions are shown for the case of TOPS and Pioneer in Fig. II-21 and II-22, respectively. Short-duration missions, ∿1.5 hr, using a different scheme for the probe/spacecraft approach geometry than that used in the parametric study, provide a simpler tracking antenna arrangement (±45 deg scan requirement) and are examined for use with the Pioneer in Mission B of Section E.

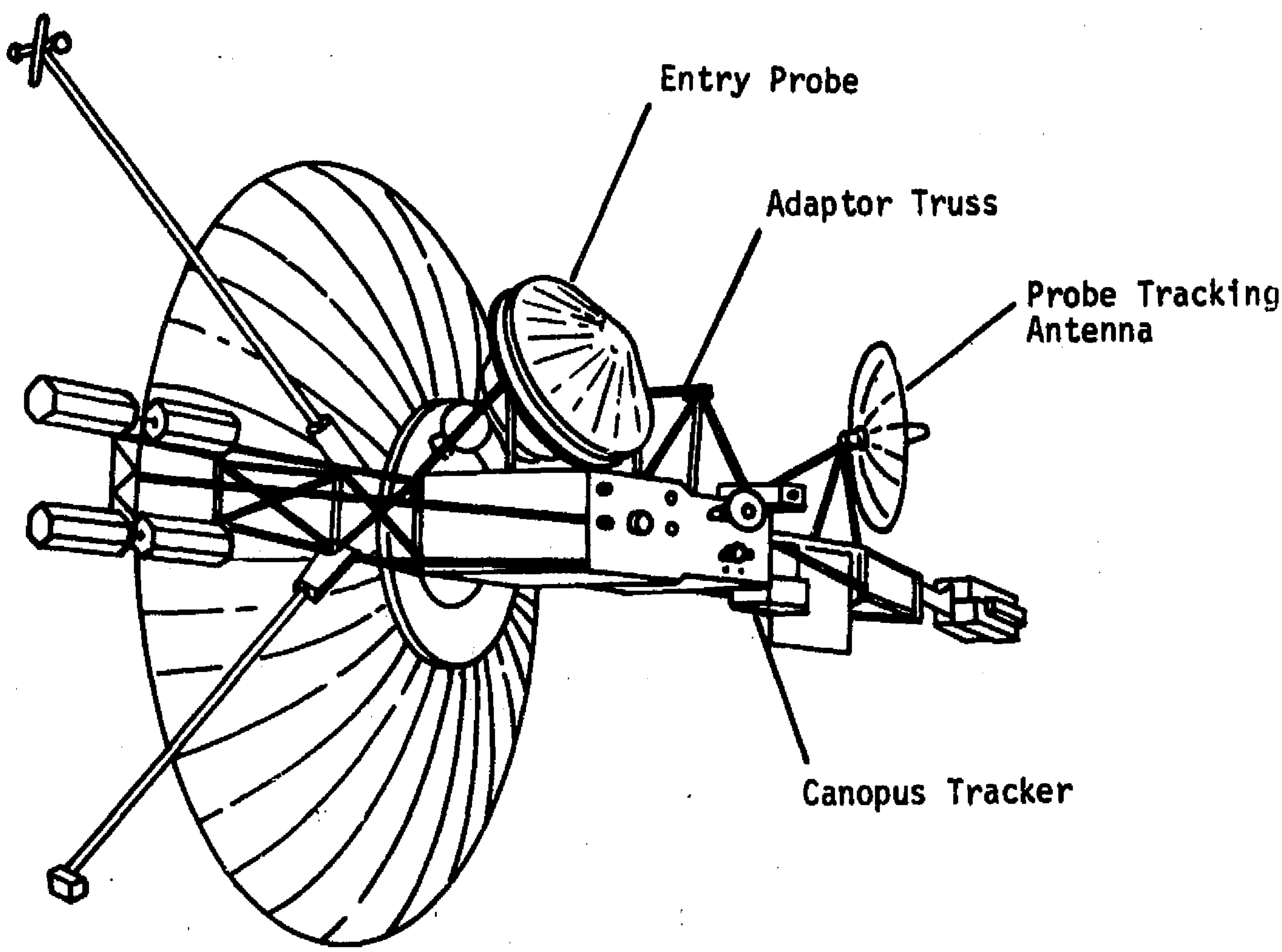


Fig. II-21 TOPS Planetary Vehicle

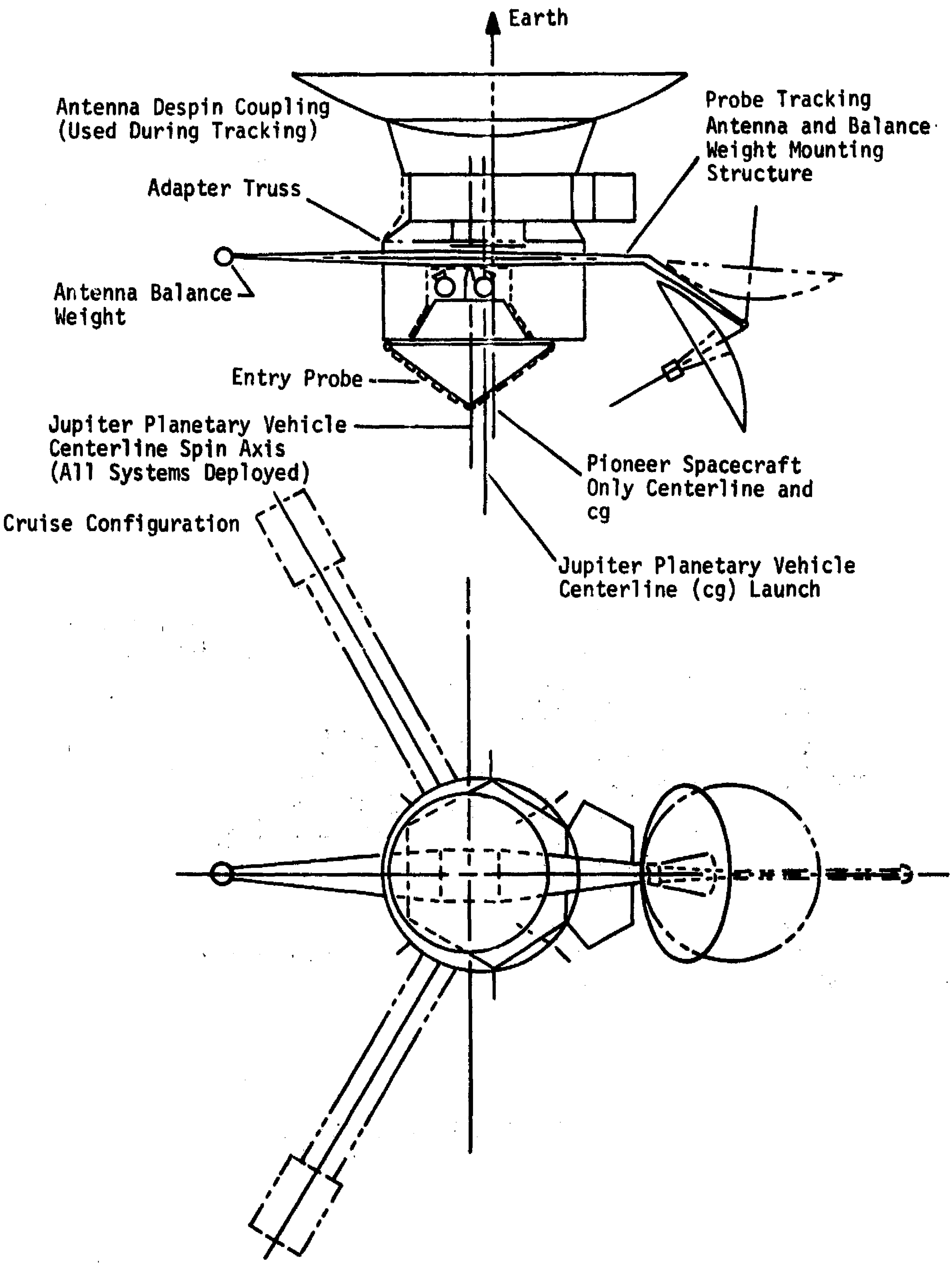


Fig. II-22 Planetary Vehicle Configuration, Pioneer F/G Spacecraft

E. DESIGN EXAMPLE AND SAMPLE MISSION SUMMARY

The parametric studies described in Section D provide a general insight into the sensitivity of probe systems to the significant mission parameters. From the range of parameters studied, one specific mission was established on which to base a more detailed design study. This was designated the Design Example Mission. In it a TOPS spacecraft is used to deliver and relay data from a probe designed for descent to 300 atm with the nominal science payload. Launch occurs in 1978 using the 7-segment version of the Titan IIID/Centaur/Burner II launch vehicle. The 300 atm pressure depth is about the maximum practical depth for a single probe transmitting data directly to the spacecraft with the given science package due to increasing transmitter power requirements with increasing depth. It should be emphasized that this design example mission is only an example and that many other viable missions of the "design example" class could be conceived involving different science payloads and descent depths.

In addition to the entry-probe-emphasis type of mission represented by the design example mission, other missions were selected to illustrate other aspects of the exploration of Jupiter with entry probes. Designated as Sample Missions A through F, these missions are listed below:

<u>Mission A</u> - A nominal science probe in combination with a 1979 Grand Tour Jupiter/Uranus/Neptune mission;

<u>Mission B</u> - A mission designed to simplify the spin-stabilized Pioneer tracking antenna implementation by selection of a reduced spacecraft-to-probe aspect angle geometry;

Mission C - A probe mission in conjunction with a Pioneer Orbiter mission;

Mission D - Two miniprobes combined with a 1979 Grand Tour Jupiter/Uranus/Neptune. One to go to Jupiter and one for Uranus (or Neptune);

Mission E - A probe mission in combination with a 1978 Grand Tour Jupiter/Uranus/Neptune mission;

Mission F - A direct link probe.

1. Comparative Data for the Design Example and Sample Missions

The tables and figures on the next few pages provide an overview of the design example and sample missions and compare design parameters for corresponding mission phases.

a. Entry and Descent - Table II-8 defines the science payloads for collecting data while descending through the Jupiter atmosphere. Descent profiles of pressure vs time are shown in Fig. II-23. Weights and other performance data for the entry and descent phases are tabulated in Table II-9. Inboard profiles of entry probes capable of performing these missions are shown in Fig. II-24.

b. Planetary Encounter - Traces and quantitative descriptions of the trajectories of the spacecraft and entry probes in the vicinity of Jupiter are shown in Fig. II-25.

c. Launch and Interplanetary Cruise - Parameters showing areas of commonality and differences for launch and interplanetary cruise for the different missions are tabulated in Table II-10, while launch configurations using the TOPS spacecraft, and also the Pioneer F/G are shown in Fig. II-26.

d. Sequence of Events - Table II-11 lists the sequence of events common to most missions.

A brief discussion of the design example and sample missions follows with more detailed information provided in Volume II.

Table II-8 Summary of Science Payloads

	Design Example, Single Probe	Mission A			Mission B, Single Probe	Mission C, Single Probe	Mission D, Single Probe	Mission E, Single Probe	Mission F, Single Probe
		Single Probe	Split						
			Upper	Lower					
1. GC/MS	x	x[a]	x[a]	x[a]	x[a]	x[a]	x[b]	x	x[b]
2. Pressure	x	x	x	x	x	x	x	x	x
3. Temperature	x	x	x	x	x	x	x	x	x
4. Visual Photometers	x[c]				x[d]	x[d]			
5. Accelerometers	x	x	x	x[f]	x	x	x[f]	x	x[f]
6. Nephelometer[e]		x	x		x	x	x	x	x
7. IR Radiometers (5 μ & 10 μ) Down-looking					x	x		x	
8. RF Lightning Detector and Microphone					x	x		x	

[a]Expanded weight GC/MS.

[b]1 to 5 amu MS only.

[c]6 channel.

[d]3 channel.

[e]3 color filter.

[f]Used during entry only; others are for turbulence.

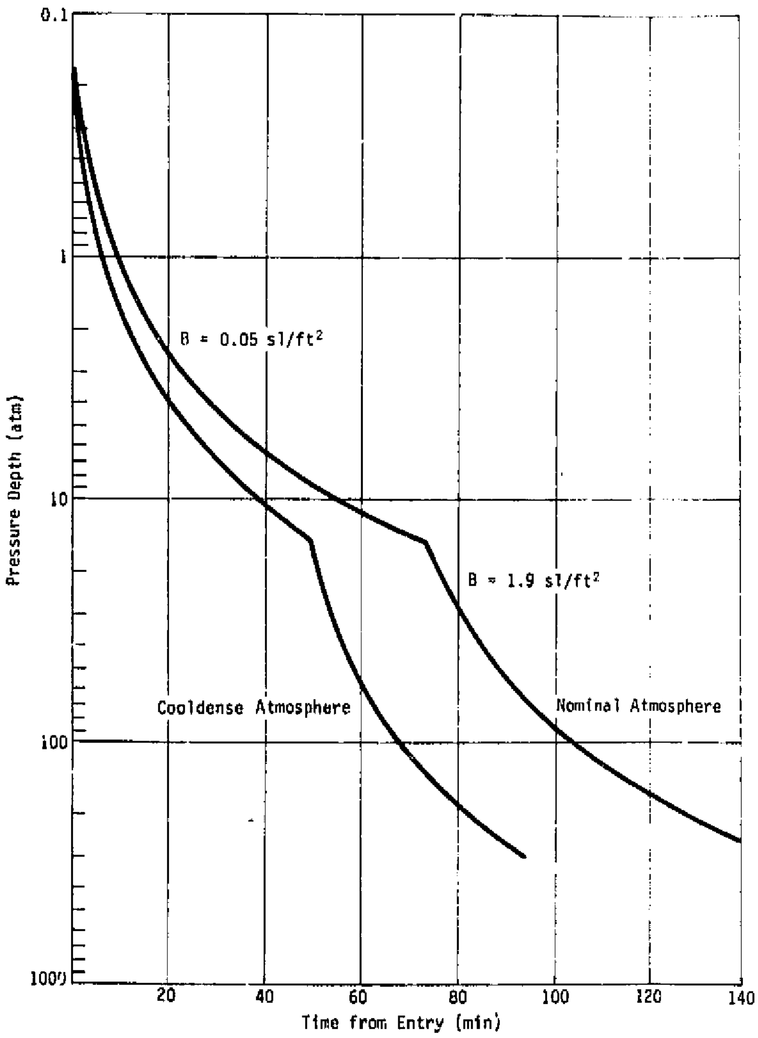
Table II-9 Summary of Entry and Descent Characteristics, Design Example and Sample Missions

Characteristic	Design Example	Mission A_1	Mission A_2	Mission B	Mission C	Mission D	Mission E	Mission F
Spacecraft Mission Δ	Flyby	J_PUN	J_PUN	Flyby	Orbiter	J_PU_PN	J_PUN	Flyby
Spacecraft Periapsis Radius, $R_{♃}$	2	6.8	6.8	2.0	2.0	6.8	1.6	2.0
Data Return Mode	Relay	Relay	Relay	Relay	Relay	Relay	Relay	Direct Link
Spacecraft Antenna Diameter (ft)	4.2	4.5	4.5	3.5	3.5	4.14	3.4	
Probe Type*	Single Staged	Single Unstaged	Split Unstaged	Single Staged	Single Staged	Single Unstaged	Single Staged	Single Unstaged
Entry Angle (deg)	-20	-15	-15	-30	-30	-15	-10	-49
Entry Weight (lb)	427	239	367	457	457	150.5	353	497
Entry Ballistic Coefficient (sl/ft^2)	1.03	0.78	0.88	1.09	1.09	0.71	0.84	0.80
Entry Vehicle Diameter (ft)	3.5	3.0	3.5	3.5	3.5	2.5	3.5	4.0
Bit Rate Preentry (bps)	180			100	100		180	
Bit Rate Descent (bps)	40/10	40	40	40	40	20	40/20	20
Probe Transmitter Power (watts)	40	40	40 Upper 2 Lower	40	40	40	40	15
Science Payload (lb)	19	10	10	27	27	10	26	10
Maximum Depth (atm)	300	10	100	73	73	17	100	17
Descent Time† (hr)	2.47	0.93	0.93	1.5	1.5	.93	1.65	.93

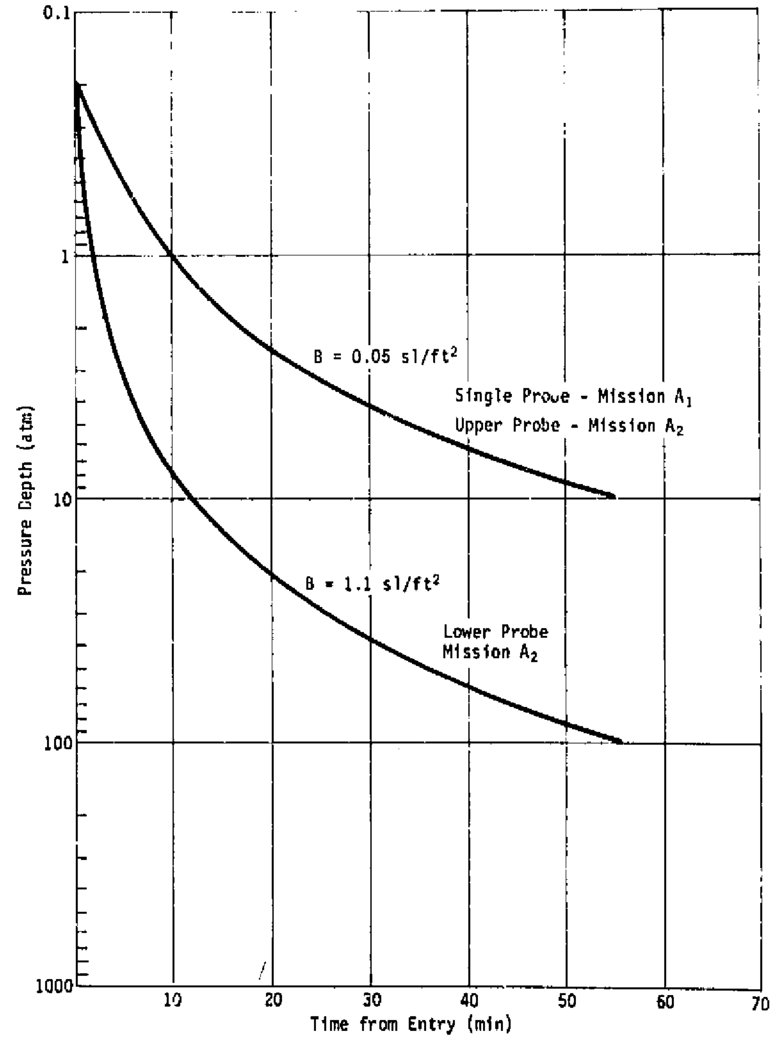
*A staged probe uses a parachute during part of the descent to shape the descent profile.

†From time of aeroshell staging.

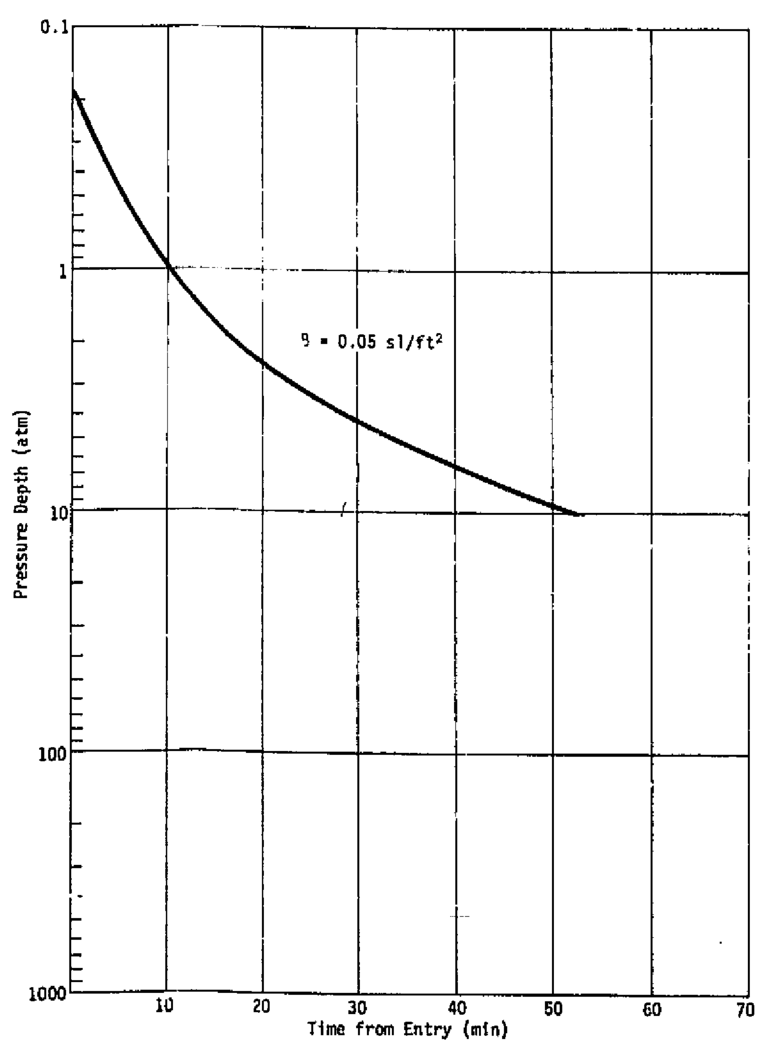
ΔSubscript P denotes a probe to that planet on a Jupiter/Uranus/Neptune mission.



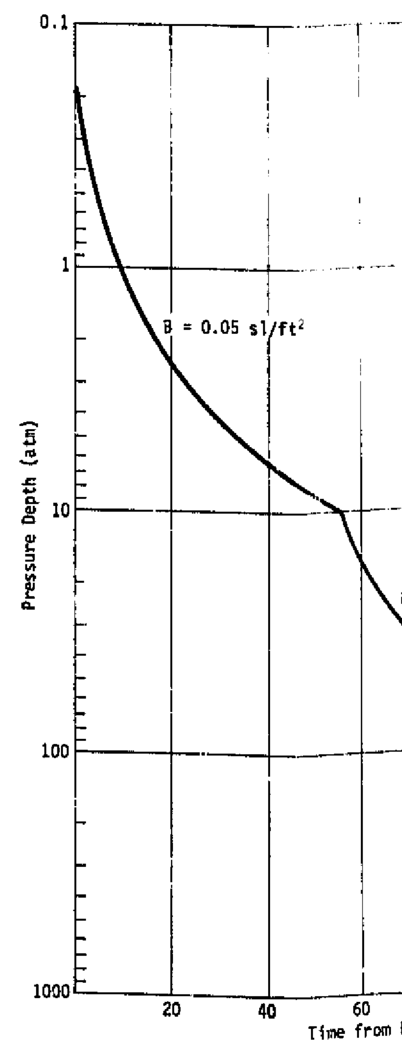
(a) Design Example Mission



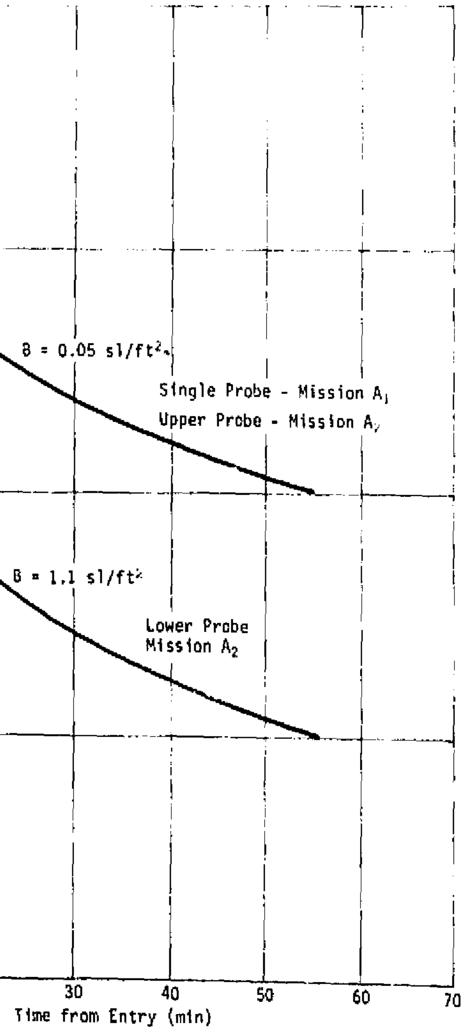
(b) Sample Mission A



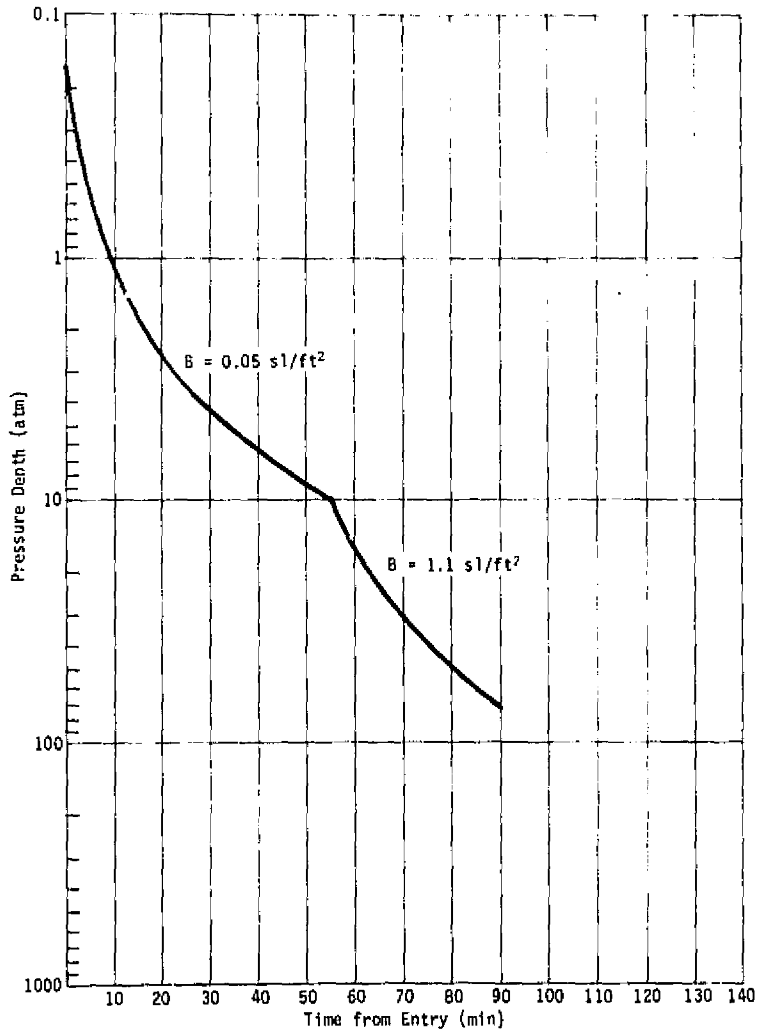
(d) Sample Missions D and F



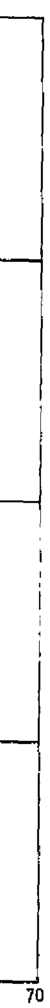
(e) Sample

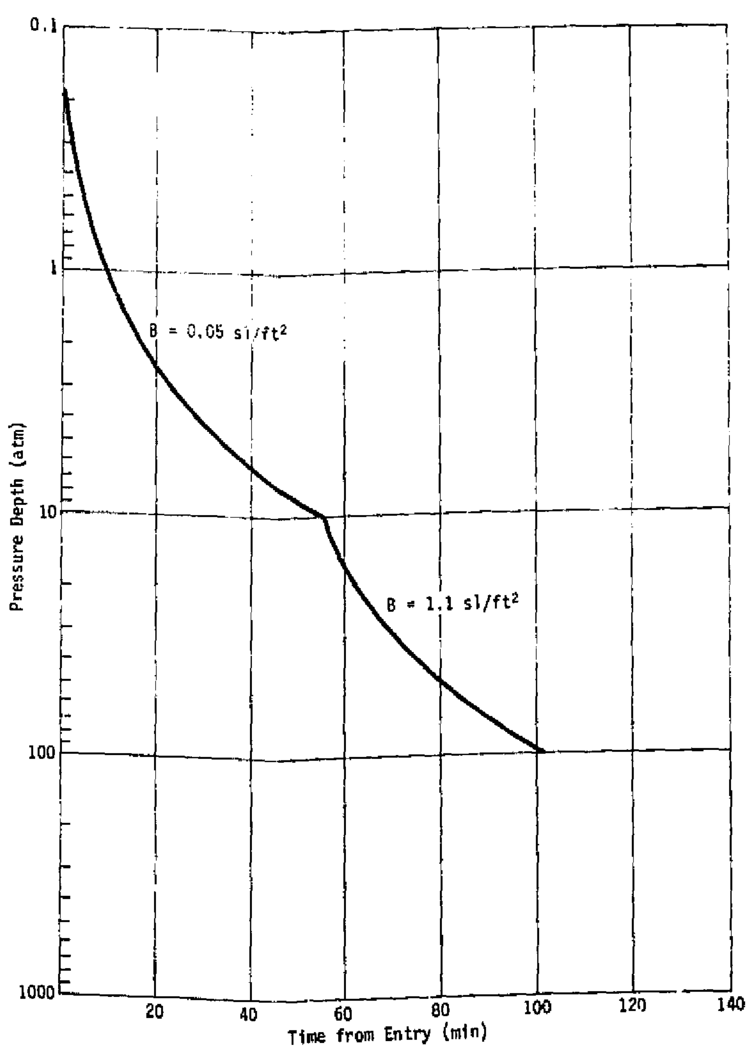


(b) Sample Mission A



(c) Sample Missions B and C





(e) Sample Mission E

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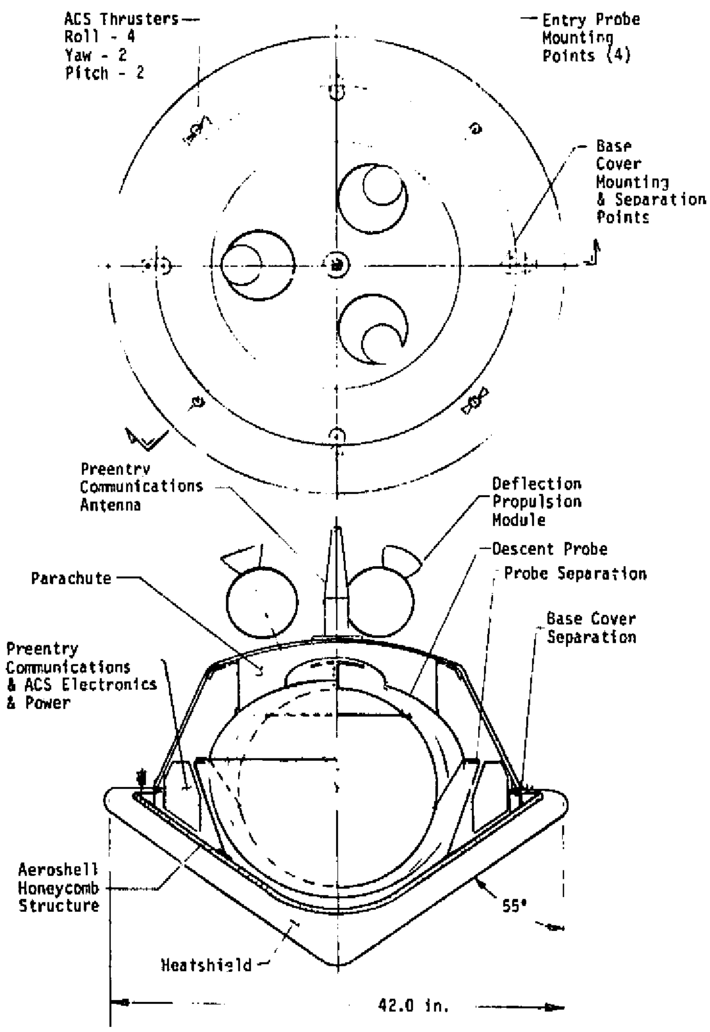
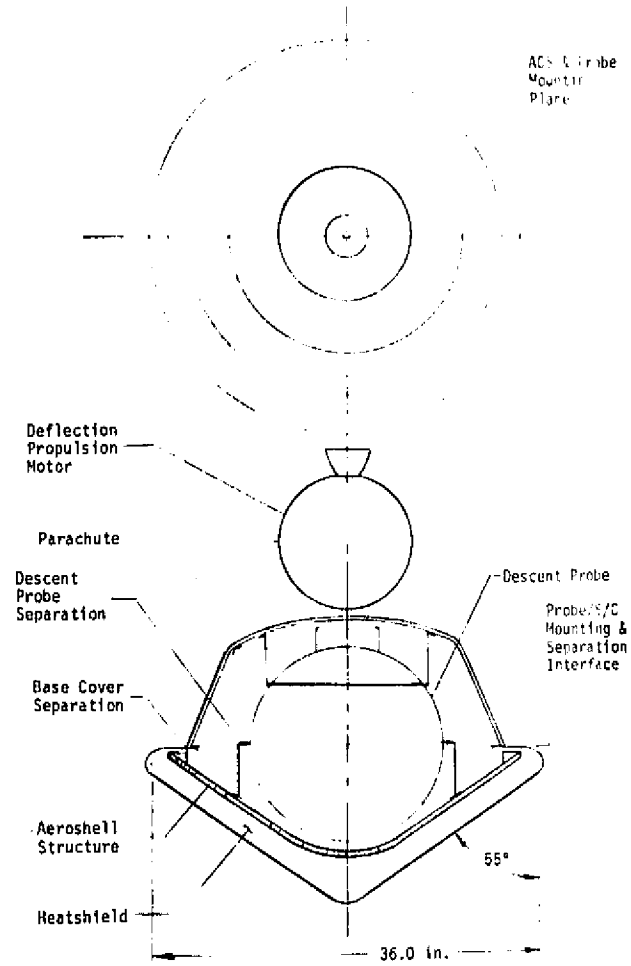
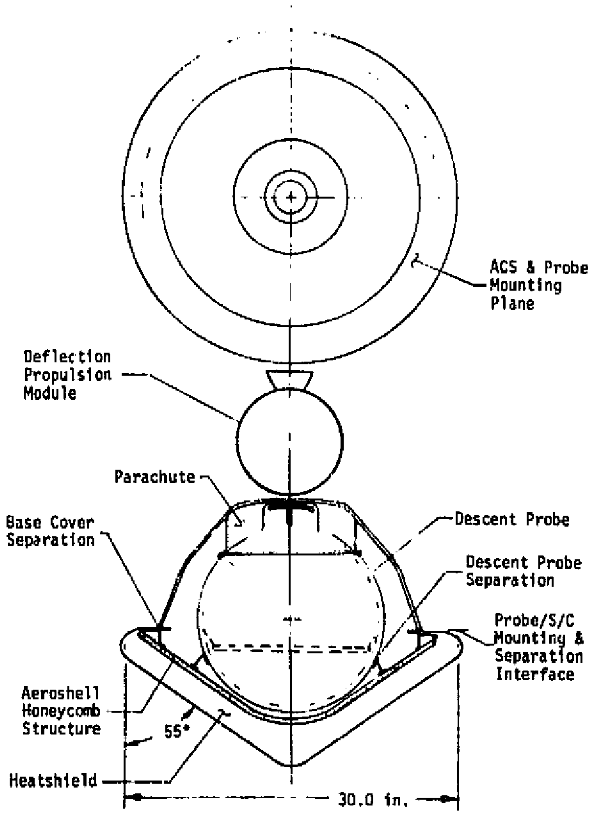
Fig. 11-23 Descent Profiles



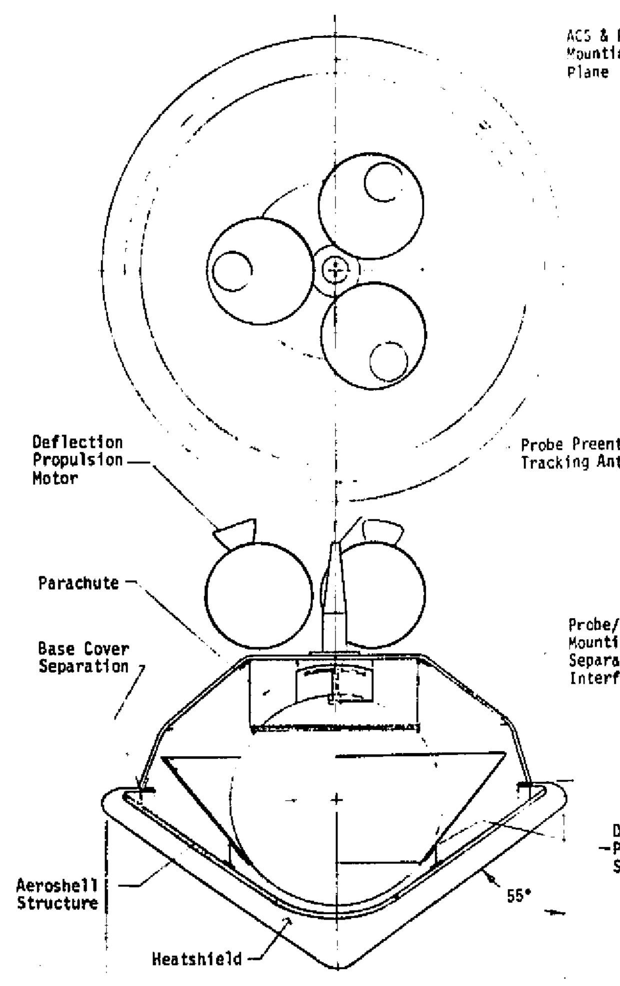
(a) Design Example



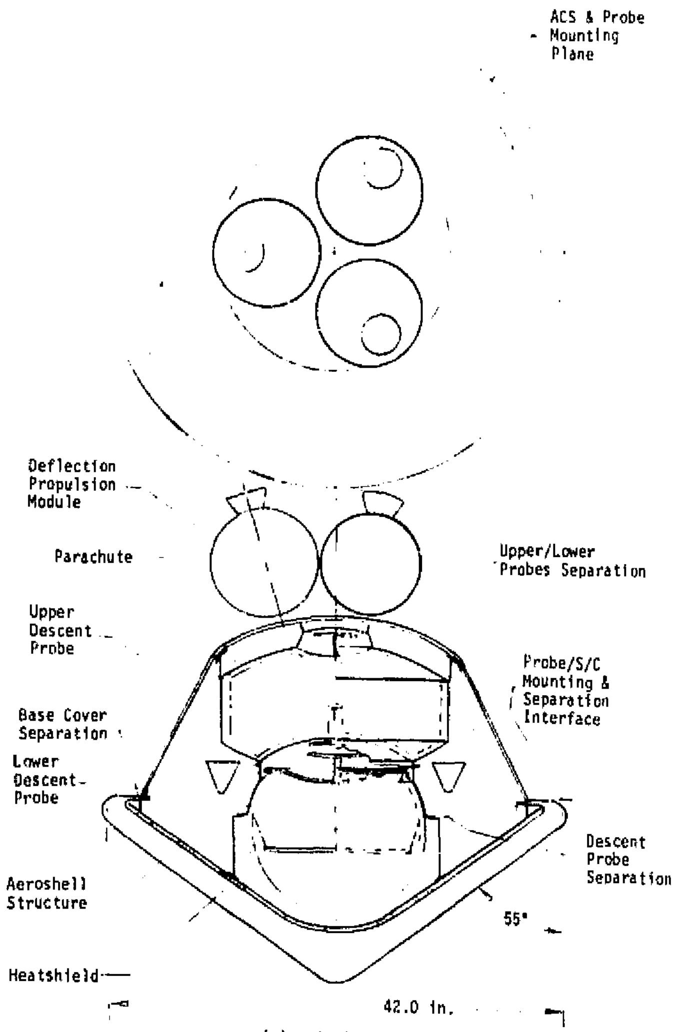
(b) Mission A_1



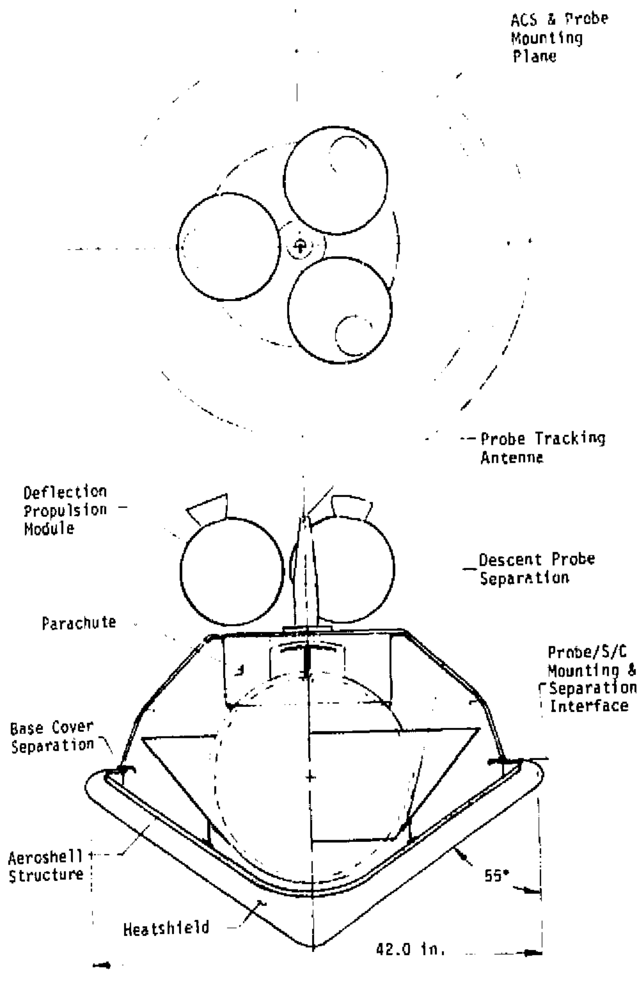
(e) Mission D



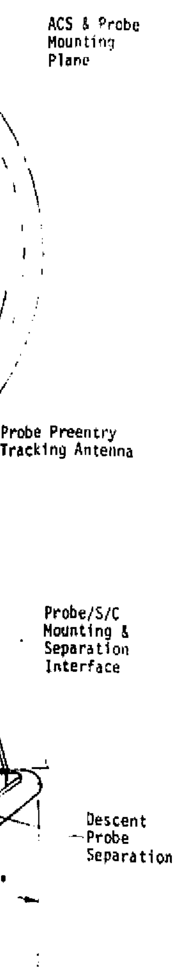
(f) Mission E

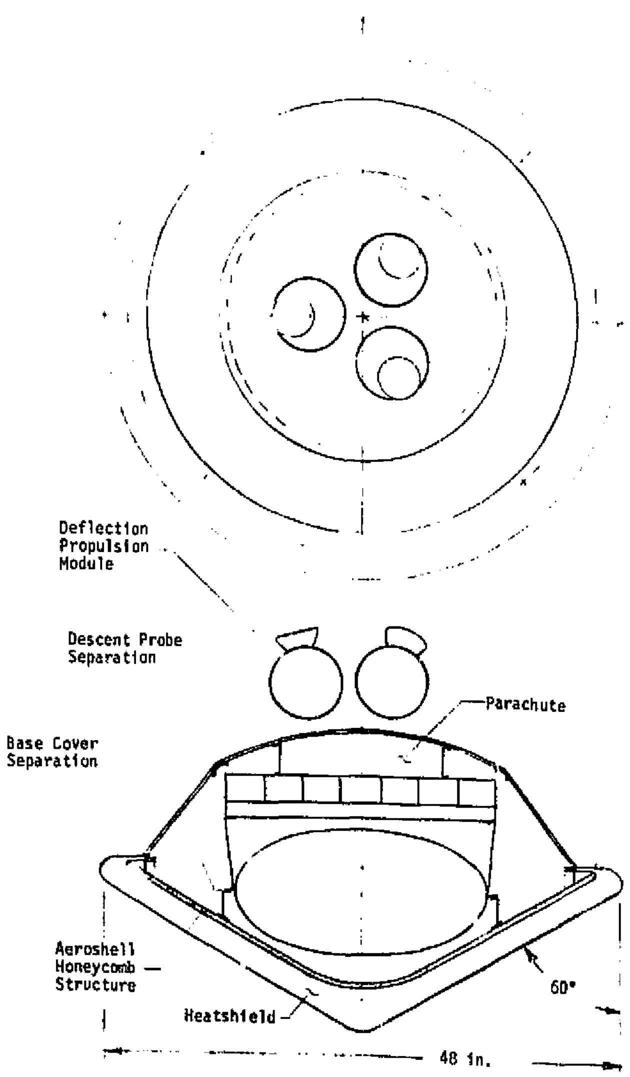


(c) Mission A_2

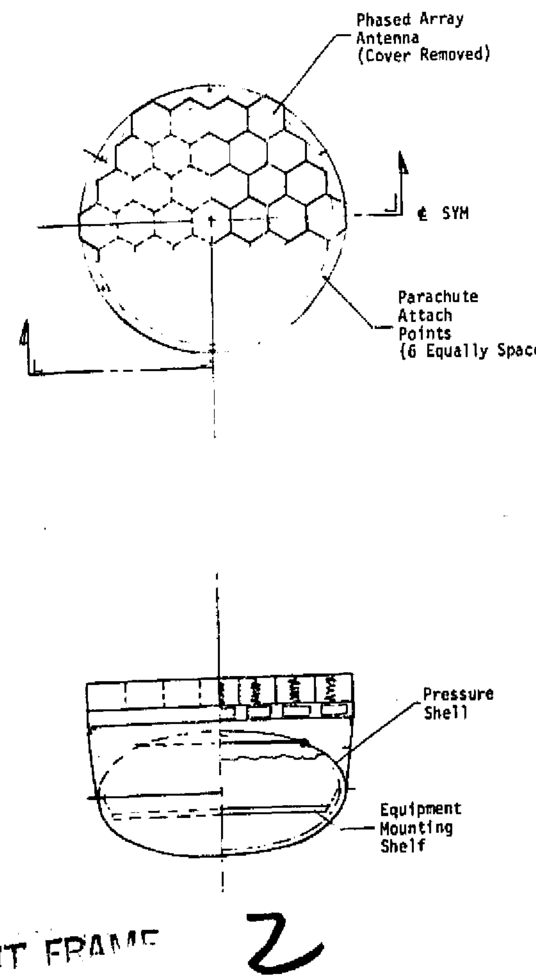


(d) Missions B and C





(g) Mission F



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(h) Mission F, Descent Probe

Fig. II-24 Entry Vehicle Inboard Pro

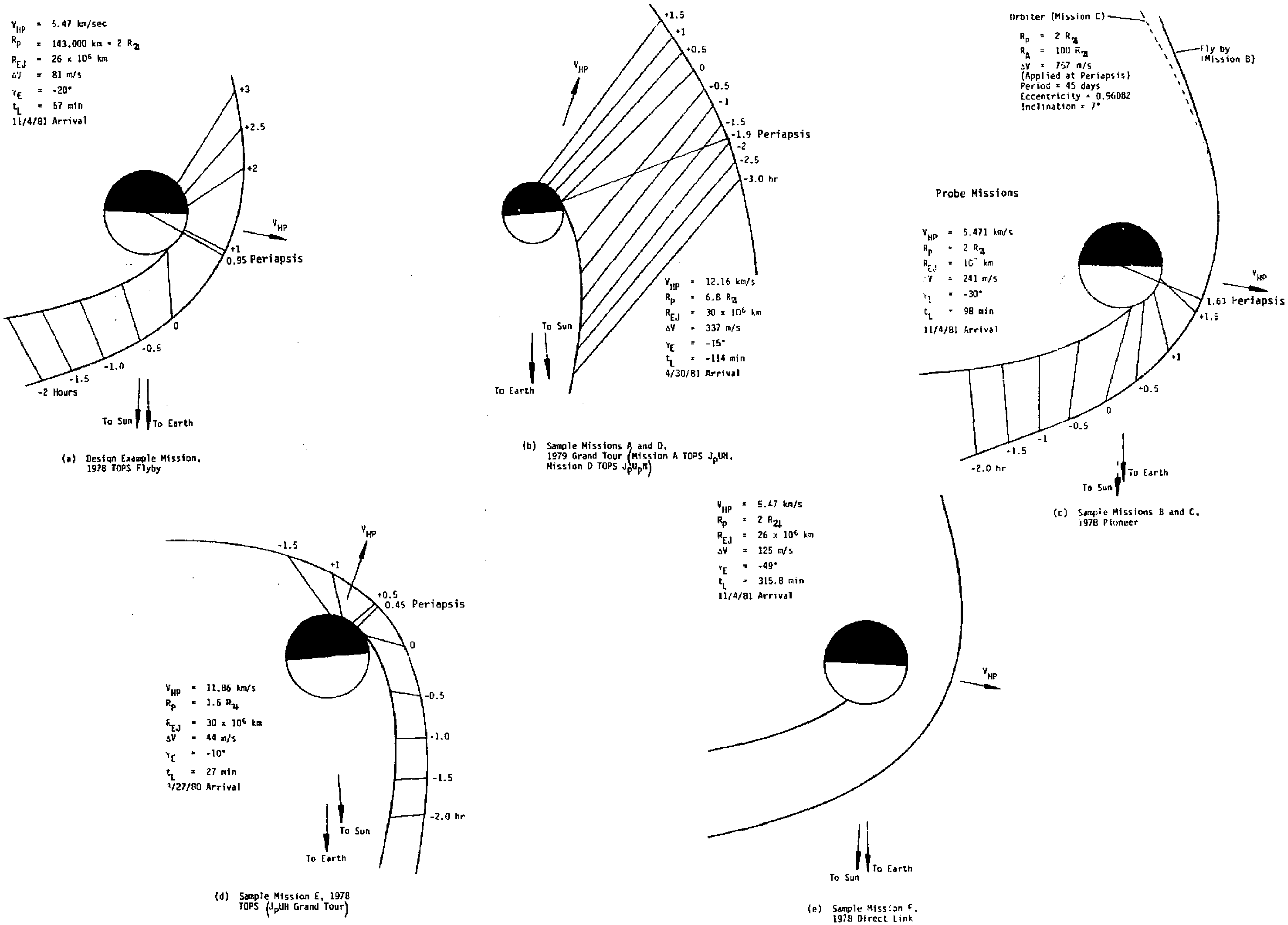
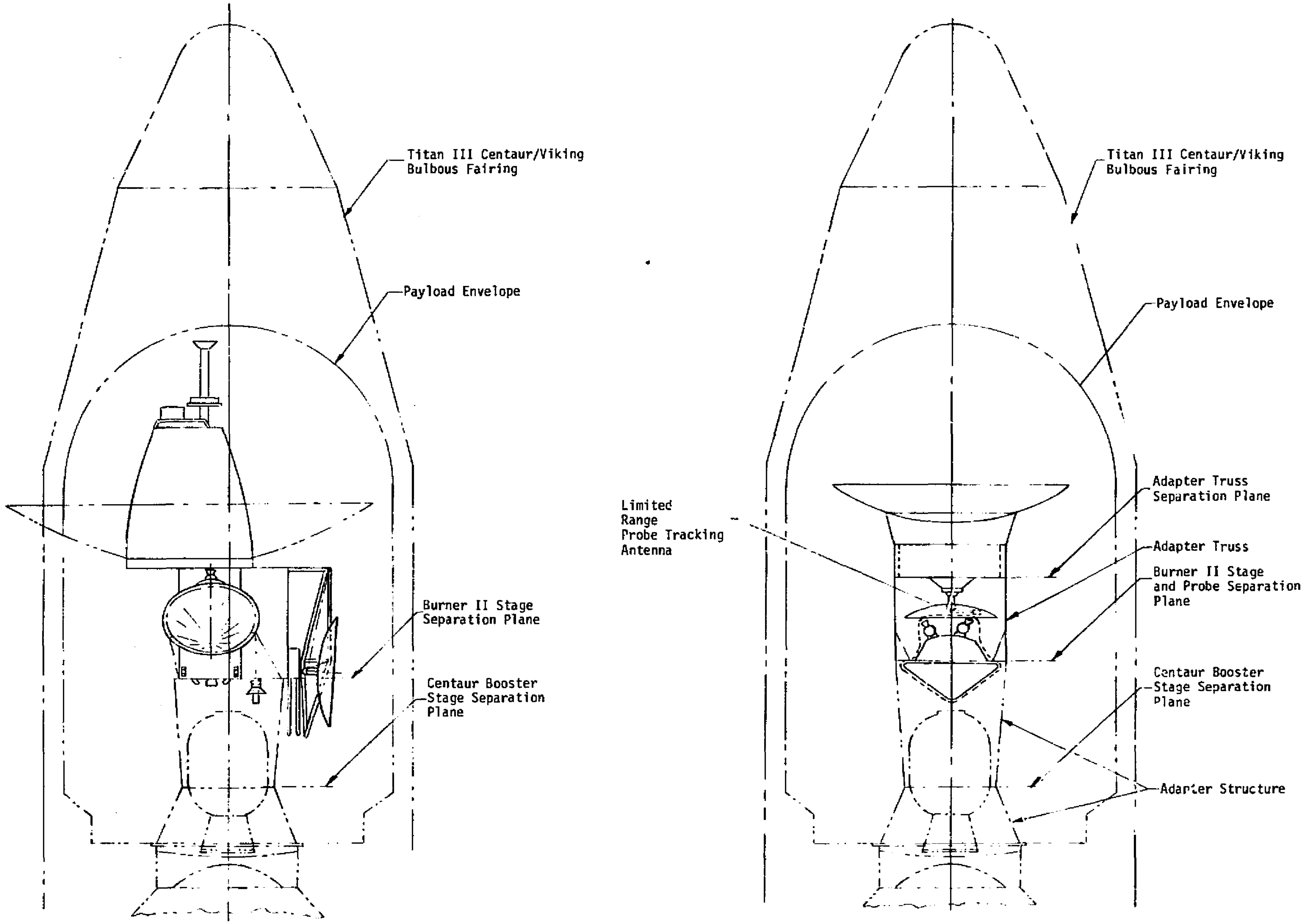


Fig. II-25 Approach Geometries

Table II-10 Summary of Launch and Interplanetary Characteristics, Design Example and Sample Missions

Characteristic	Design Example	Mission A_1	Mission A_2	Mission B	Mission C	Mission D	Mission E	Mission F
Spacecraft	TOPS	TOPS	TOPS	PIONEER	PIONEER	TOPS	TOPS	TOPS
Spacecraft Mission	Flyby	J_pUN	J_pUN	Flyby	Orbiter	J_pU_pN	J_pUN	Flyby
Launch Date	9/25/78	11/6/79	11/6/79	9/25/78	9/25/78	11/6/79	10/10/78	2/25/78
T III D/Centaur/Burner II (SRM Segments)	7	7	7	5	5	7	7	7
Launch Capability, Payload (lb)	2420	1900	1900	1550	1550	1900	1900	2420
Liftoff Weight, Payload (lb)	2019	1849.5	2012.5	1300	1600	1900	1918	2067
Probe System Weight (incl S/C Mods)(lb)	569	399.5	562.5	743	1043	450	468	617
Probe Installed Weight (lb)	467	239	367	552	552	185	381	552
Injection Energy, C_3 (km^2/sec^2)	99	112	112	86.2	86.2	112	112.4	99
Trajectory Type	II	I	I	II	II	I	I	II
Arrival Date	11/4/81	4/30/81	4/30/81	11/4/81	11/4/81	4/30/81	3/27/80	11/4/81



(a) TOPS (b) Pioneer

Fig. II-26 Planetary Vehicle/Booster Installations

Table II-11 Sequence of Events

Turn On, Check Out, and Update Entry Vehicle for Separation
Orient Entry Vehicle for Deflection Impulse
Separate Entry Vehicle at 0.3 mps and Stabilize
Deflect Entry Vehicle to Entry Trajectory $\left(R_{EJ} = 2.6 \times 10^7 \text{ km}, \Delta V = 81 \text{ mps}, \tau = - 106.5 \text{ deg}\right)$
Orient to Zero Angle-of-Attack at Entry
Spinup for Stabilization during Coast (ω = 3 rad/sec)
Coast to the Planet ($\sim$ 40 Days)
Establish Preentry Communications Link ($R = 3 R_{♃}$, Entry - 1.2 hr
Acquire Preentry Science (Entry - 1 hr)
Activate Ion Mass Spectrometer
Despin prior to Entry (Entry - 17 minutes)
Entry (Entry Angle - 20 deg)
Sense 0.1g for Reference Timing (Entry + 10 sec)
Reestablish Communications Link after Blackout
Deploy Parachute, Stage Aeroshell/Heat Shield, Activate Descent Science (Timed Function) (Entry + 62 sec)
Change Data Modes and Rates (Pressure-Sensed Functions, P = 30 atm)
Stage Parachutes (Where Required)
Complete Descent Mission
<u>Note</u>: Prior to start of the sequence, final trajectory corrections have been made by the spacecraft at entry - 40 days.

2. Design Example Mission

This mission examines the situation for a probe-optimized mission using the flyby TOPS spacecraft as a relay link. One hour of lightside science data is provided with the Type II trajectory at a relatively shallow entry angle, -20 deg. This angle is large enough to assure no skip out, even with less than ideal ΔV implementation accuracy, and is small enough not to impose large entry aeroshell weight penalties. The sum of the entry and thermal/structural descent protection systems amounts to a total of 53% of the probe entry weight of 427 lb. Another 40 lb are required to implement the probe deflection operation, and 102 lb for adapting the 3.5-ft-diameter probe and the 4.2-ft-diameter autotracking antenna to the TOPS spacecraft, 569 lb altogether.

The resulting total liftoff weight of 2019 lb is within the 2420-lb capability of the 7-segment Titan IIID Centaur launch vehicle, with Burner II, and also the 5-segment version with the stretched Centaur.

Data return from the nominal science payload, 40 bps for the first 1-3/4 hr and 10 bps for the final 3/4 hr descent to 300 atm requires a 40 watt solid-state transmitter. Power is provided by a remotely activated zinc-silver primary battery.

The 300 atm depth is deep enough to get below the water clouds even in the cool model atmosphere. However to accomplish this mission in the cool-dense model atmosphere as well as the nominal would require a 50% increase in transmitter power and a 20% increase in antenna size. The total probe system weight penalty would be 30%, 740 lb vs 569 lb. This is still within the launch vehicle capability but large enough to make the nature of the spacecraft modifications more difficult.

An alternative situation, designing for 300 atm only for the case of the cool-dense atmosphere, can be accomplished for essentially the same weight as the nominal model probe. Although the resulting cool model probe would survive to only 20 atm in the nominal atmosphere, that should be deep enough to include the water cloud region.

3. Mission A

This mission was selected to examine what kind of a probe mission could be accomplished within the constraints of a Type I, 1979 Jupiter/Uranus/Neptune Grand Tour mission. Both single and split probes were evaluated.

It was found that a single probe to 10 atm with a slightly larger than nominal payload -- a nephelometer replacing the photometers for the darkside measurements -- could be delivered with a total probe system weight of 400 lb, 50 lb under the available weight for that mission. Slightly greater depths could be achieved by using the extra weight, but the transmitter power, already at 40 watts, would then become critical.

In an attempt to improve the depth situation, a split probe to 100 atm was evaluated (Mission A_2) and found to be somewhat over the limiting payload weight, 560 vs 450 lb. Reducing the 100 atm depth to a shallower value was found not to improve the weight picture significantly, but increasing the R_p of the swing-by a small amount (from 6.8 to 7.8 $R_{♃}$) was found to increase the payload capability to the required value without significantly altering the split probe power requirement or weight (see Volume II, Chapter V).

4. Mission B

Mission B was designed to see what could be accomplished with a small range of tracking angle coverage. This arrangement was devised for the Pioneer spacecraft in view of the difficulty of obtaining wide coverage from the spinning spacecraft.

The science payload was enlarged from 16 to 27 lb, but the probe depth was found to be constrained to ∿70 atm by the reduced time associated with the limited (±45 deg off-axis) coverage capability identified for the tracking antenna. The resulting entry weight of this 70 atm probe, 457 lb, is comparable to the 300 atm design example probe, 427 lb, i.e., the steeper entry angle (required by the ±45 deg coverage geometry) and the larger science package combine to offset the weight gained by the shallower depth of penetration.

Based on the nominal atmosphere model, the Mission B probe makes more efficient use of the weight than does the design example probe, since in both cases the depth of the water clouds are exceeded and the Mission B probe carries almost twice the science payload as the design example. However, an expanded payload could also be flown on a design example class of mission. This could be accomplished in one of several ways:

1) The 300 atm descent depth could be retained and the approximately 15% increase in total probe system weight simply accepted, or
2) The depth could be reduced enough to offset the science payload expansion. A depth of about 200 atm appears to achieve this (from the parametric data of Section D).

5. Mission C

Sample Mission C was selected to demonstrate the feasibility of using the Pioneer F/G as an orbiter after completion of the atmospheric entry mission as in sample Mission B. The approach geometry for sample Mission C is the same as for sample Mission B.

The probe systems of Mission C are also identical to those for Mission B, since the probe mission is complete before orbit insertion. However, the configuration of the orbiter design must take into consideration the probe and probe tracking antenna in t e location of the deorbit propulsion system. The orbit insertion system will have to account for any added weight of leftover probe systems.

Based on a modified Pioneer weight of 550 lb --- less orbit insertion system -- the total orbit insertion system is estimated to be 300 lb. Consequently the liftoff weight of Mission B, 1300 lb, would increase to at least 1600 lb, exceeding the 1550 lb capability of the 5-segment TIIID Centaur with Burner II, but still falling within the capability of the larger launch vehicles.

6. Mission D

Mission D was selected to explore the possibility of a small enough probe to permit the use of dual probes -- one at Jupiter and one at a subsequent planet -- in connection with a Grand Tour mission.

By reducing the science payload (see Table II-8) it was found that a 20 bps return from a single probe descending to 17 atm could be accomplished for a total entry weight of 150 lb. Two such probes can just be carried on a 1979 JUN mission.

The data return from this lighter weight probe does not warrant its use for a single planet mission. However the mission does demonstrate the feasibility of conveying comparable probes to more than one planet. The actual allocation of science payloads, and consequent weight, between planets should be based on optimization of overall science value.

7. Mission E

Since the large radius of periapsis, 6.8 $R_{♃}$, of the 1979 Grand Tour mission was found to limit probe performance (see Mission A), a 1978 Grand Tour mission was examined to determine the influence of a small R_p on the probe systems. The 1.6 $R_{♃}$ for the 1978 mission was found also to present relay communications link constraints but of lesser magnitude than the 1979 case. Mission time is limited to 1-1/2 hr by the increasing probe/spacecraft aspect angles (due to the rapidly changing relative geometry caused by the small R_p). However even with an expanded science payload, 26 lb, the Mission E single probe can descend deeper, 100 atm vs 10 atm, than does the 1979, Mission A, probe.

The total system weight for the Mission E conditions was found to be slightly over the launch vehicle capability, 469 lb vs 450 lb, indicating that slight reductions in depth or science payload would be required.

8. Mission F

Sample Mission F was selected to examine the feasibility of a direct radio link to earth. The TOPS spacecraft is used and its flight path is similar to that of the design example mission. The probe enters at -49 deg and the descending capsule goes to a depth of approximately 20 atm. A 10-lb science payload identical to that for Mission D provides 20 bps with a high gain (22 dB) self-focusing array antenna radiating 15 watts RF power. The array weighs 51 lb and this in combination with the steep entry necessitated by targeting in the subearth region results in heavy probe, 497 lb at entry.

Little possibility exists for extending the performance level of direct link missions much further than Mission F due to the following constraints.

1) The mission time, and hence depth, is limited by the rapid rotation of the planet; the probe moves away from the subearth region;
2) The entry angles required to get steeper as the mission duration is increased;
3) Type I trajectories require even steeper entry angles to achieve the subearth targeting.

F. ENVIRONMENTAL UNCERTAINTIES: PRESENT AND FUTURE

Of all the various environments that the probe must pass through and survive during the mission, the most hostile and uncertain are the asteroid belt, the Jovian radiation belt, and the atmospheric entry and descent environments. The asteroid environment was only briefly considered in this study and the probability of surviving it is to be investigated in part by Pioneer F/G. The radiation belt environment was considered and it appears that it is a straightforward design problem to shield against the nominal model. Pioneer F/G will probably reduce the uncertainties in the radiation belt models if they survive the asteroid belt. Survival of the entry heating environment was an assumption of the study since this problem is not yet amenable to a theoretical treatment and probably never will be amenable to earth laboratory investigations short of exposure to a nuclear device. However, the uncertainties and penalties associated with designing to a wide range of upper atmosphere conditions will be reduced by the Pioneer F/G occultation experiments (if they survive the asteroid and radiation belts).

Thus the Pioneer F/G missions may reduce the environmental uncertainties and hence, the associated probe mission design penalties. However, unless the Pioneer F/G missions attempt to complete occultation (to critical refraction), very little will be learned about the lower atmosphere. A complete occultation would not reduce all the lower atmosphere uncertainties but it would distinguish between the cool model for which critical refraction occurs near 2 to 3 atm and the nominal model for which it occurs near 20 atm. Possibly, some information on the NH_3 and H_2O abundances and the turbulence could also be obtained, but would be somewhat model dependent.

Shallow or grazing occultations of Pioneer F/G could reduce the uncertainties in the upper atmosphere scale heights, the mean molecular mass, and the pressure-temperature near the cloud tops. However, the present wide range of lower atmosphere pressure-temperature profiles, and uncertainties in the NH_3 and H_2O abundances, would remain. A complete occultation might at least reduce the range of possible pressure-temperature profiles. It appears that the uncertainties in the NH_3 and H_2O abundances and the turbulence will not be resolved until a probe actually enters and survives to below the H_2O cloud.

G. SYSTEM IMPLEMENTATION STRENGTHS AND WEAKNESSES

1. Navigation System and Accuracy

The accuracy analysis indicates that the probe deflection maneuver errors are the dominant source of dispersions in the critical mission parameters. Navigation errors have little effect on the missions considered. Dispersons in the probe trajectory due to deflection system errors influence the missions in two major areas. Probe entry flight path angle dispersions require the design entry flight path angle to be steep enough to avoid skipout. The second area is the probe/spacecraft communication geometry.

Accuracy analyses were conducted using Monte Carlo techniques to obtain the dispersions in the probe entry parameters, communications geometry, and communications power requirements. Three sources of error were evaluated: probe deflection maneuver errors, uncertainty in the 1 atm level of the Jovian atmosphere, and uncertainty in the Jovian gravitational constant. The gravitational constant uncertainty was found to have little effect and can be neglected.

Dispersions in the entry flight path angle determine the shallowest possible entry angle. These dispersions are a function of the level of implementation errors, the spacecraft periapsis radius, and the nominal entry flight path angle. Steeper entry flight path angles exhibit less sensitivity to implementation errors than the shallow entry cases.

Entry flight path angle dispersions increase as the periapsis radius of the spacecraft increases. For a spacecraft periapsis radius of 6.8 Jupiter radii, the shallowest allowable entry angle is about -15 deg for deflection errors of 1% and 1 deg (3σ). For a periapsis radius of 2 Jupiter radii, the shallowest angle is about -8 deg. The probe deflection radius has little effect on the probe entry flight angle dispersions.

From the standpoint of entry flight path angle dispersions, deflection maneuver implementation errors up to 4.5% and 4.5 deg (3σ) can be accommodated. For alternate Mission A the nominal entry flight path angle would have to be about -30 deg instead of -15 deg to accommodate the larger errors. The design entry angles for alternate Mission B and the design example mission are acceptable for these larger deflection errors.

For shallow entry flight path angles, the 1% and 1 deg (3σ) deflection system errors produce about the same flight path angle dispersion as a 1000 km (3σ) uncertainty in the radius of the atmosphere. As the entry angle becomes steeper, the atmospheric radius errors are less important.

Probe/spacecraft communication is the aspect of these missions most sensitive to the sources of error considered. The first part of the problem is pointing the spacecraft antenna at the probe. Spacecraft antenna pointing is prescribed by cone and clock angles referenced to earth and Canopus. The design example mission requires the probe to be acquired by the spacecraft at approximately an hour before entry. The dispersions in the required cone and clock angles during this time are three to four times larger than the spacecraft antenna beamwidth so a search procedure must be employed to acquire the probe. The cone and clock angle dispersions are much larger for increased deflection maneuver (4.5% and 4.5 deg) errors. The probe acquisition problem is further complicated by dispersions in the probe/spacecraft range rate and range acceleration. After the probe has been acquired it must be tracked. The pointing angle dispersions for deflection system errors of 1% and 1 deg (3σ) are too large to permit preprogrammed tracking so an automatic tracking system is required.

The second part of the probe/spacecraft communication problem is the communications power loss dispersion due to probe aspect and range dispersions. During preentry and for about half of the descent phase of the design example mission, the power loss dispersions in decibels are approximately proportional to the deflection maneuver implementation errors. Power loss dispersions as large as 3 db (3σ) are experienced for deflection maneuver errors of 1% and 1 deg (3σ). Therefore, the communication power penalty in watts is very sensitive to deflection errors.

Dispersions in the spacecraft antenna pointing angles and power loss depend almost entirely on probe deflection errors. The navigation uncertainties, atmospheric radius uncertainty, and gravitational constant uncertainty have little effect. The targeted entry flight path angle also has little effect on the communications dispersions.

The analyses deal primarily with probe deflection from a flyby spacecraft. Alternatively, the spacecraft could be deflected from the impacting probe path. The major design problem, communications system dispersions, would not be significantly improved by this technique unless a program of tracking and path correction of the spacecraft from deflection to probe entry could be instituted. In the design example mission such an approach was not considered because the spacecraft is not visible to the earth during this period. Deflection maneuver requirements and implementation accuracies are comparable for either probe or spacecraft deflection.

2. Data Transfer System

a. Relay Links - Telecommunications link designs using probe-to-spacecraft relay are practical for both the nominal and the cool-dense atmospheres. Depths of 1000 atm can be reached by use of the split probe concept, *optimization of relay communications geometry*, and data rate switching to counteract the attenuation of radio signals.

For the Grand Tour missions, such as the 1979 JUN, optimization of communications geometry is not an option since the periapsis radius ($6.8\ R_{♃}$) is fixed by the Grand Tour objectives. The 1978 Grand Tour with $R_p = 1.6\ R_{♃}$ permits a more effective communication system design than the 1979 Grand Tour, but not so effective as with $R_p = 2.0\ R_{♃}$.

Near state of the present art has been used in all data transfer system designs with some development projecting to a 1975 technology. The proposed use of solid-state power amplifiers of up to 40 watts output at 0.85 to 2.3 GHz, a mechanically despun spacecraft antenna for Pioneer, use of complementary MOS logic, plated wire probe memory, and linear integrated circuits are all reasonable projections for 1975.

The requirement for implementing a frequency and antenna pointing search and acquisition system on the spacecraft for the relay link adds complexity since the system would be simpler were it possible to use wideband receivers and preprogrammed antenna pointing. However, since the latter approach is not practical, emphasis must be placed on developing a reliable automatic acquisition and tracking system having good search logic with immunity to false signals and on conserving weight and power in the design. The required technology is available today to develop the system; however, improvements in noise figure and size and weight would come with 1975 technology.

b. Direct Links - Missions using direct links to earth do not have probe spacecraft geometry as a consideration, but data transfer system design constrains these missions in several important respects. These include:

- High entry angles in targeting to near subearth;
- Relatively shallow depth of penetration combined with low bit rates;
- Limited aspect angles for the probe antenna over which a retrodirective array can operate.

With the exception of antenna system requirements, technological projections for the direct links are similar to those for the relay links.

3. Mechanics System

Beginning with the postentry descent phase, the major tradeoff findings are noted and the strength and weaknesses of the recommended systems discussed briefly.

a. Descent - Considering the deepest descent range, 300 to 1000 atm, the thermal/structural concept affording the lightest weight was found to be a combination of porous external insulation covering an Inconcel pressure shell and limiting its temperature to 1100°F. Multilayer insulation, located on the inner walls, further limits the temperature to 125°F and phase-change material absorbs internally generated heat.

This arrangement prevents having to operate the highly loaded structure at the ambient temperatures of 1350°F to 2100°F, but depends strongly on the performance of the external insulation for which no data exist under the design conditions. Conservative extrapolations of Min K insulation performance are used and confidence in the resulting system is high.

For the shallowest probes, 10 to 100 atm, no external insulation is required, and from 100 to 300 atm, the system performance is not so highly dependent on external insulation performance.

The pressure-resistant container design concept was selected as the basis for this study over designs that permitted the ambient pressure to penetrate into the payload compartments or designs that partially balanced the pressure with internal pressurization since no weight savings resulted. In the case of the design that allows the compartment to operate at ambient pressures, the components must take the high pressures, but the seal problems are considerably alleviated. Since preliminary indications are that many components can take the pressure, this approach deserves further examination.

b. Entry - Unknowns in the radiation heating and ablation analysis result in an uncertainty band on the heat shield weight fractions used in this study. However, the sensitivity of system performance on these weight fractions is such that somewhat greater than nominal weight fractions can be tolerated, e.g., if 50% greater weight fractions are required than those used, the added weight only reduces the 300 atm depth capability of the design example probe to 150 atm.

The structural designs of the aeroshells are based on very modest estimates of improvements in the current state of the art, and are probably conservative in terms of 1975 technology. The designs for steep entry angles are more conservative than for shallow ones since the structural configurations are optimized for the lower dynamic pressure conditions.

c. Deflection - Implementing the probe deflection maneuver from the three-axis stabilized TOPS vehicle appears to lend itself to the Viking Lander-type of guidance and control system on board the probe -- as opposed to just spin stabilization of the probe during ΔV application. This three-axis strapdown inertial guidance system will be developed prior to its requirement for Jupiter probes. It will however require further development in the area of weight minimization to attain the performance levels identified in this study.

For the case of Pioneer probes, which will be spinning upon release and which will require reorientation prior to ΔV application, a precession system on board the probe is required. This involves development of a roll reference sensor compatible with the precession system and the other onboard probe systems. Accuracy of preprogrammed turning with such a system may be less than desired and requires further evaluation.

d. Adaptation of Probe Systems to Spacecraft - Mounting of the probes to either of the spacecraft presents no major problems until the size of the probes reaches ~450 lb.

Above the 450-lb probe category, cg offsets in the case of TOPS, and roll-to-transverse-inertia ratios in the case of Pioneer, begin to create difficulties. For large probes the TOPS trajectory correction motor would have to be modified significantly since tilting its existing thrust axis as far as necessary, ~30°, would not be feasible. For Pioneer, balance weights, RTG boom extensions, or larger-than-optimum probe diameters are required with heavy probes to maintain the spin axis as the maximum moment of inertia axis.

Regarding antenna provisions; for TOPS, a relatively straightforward situation exists. The mechanism is similar to the scan platform mechanism of TOPS. For Pioneer, the despun system required has been developed, but the dynamics of its application to the Pioneer spacecraft have not been fully investigated, particularly for the boom-mounted version.

Thermal control by heater/insulation systems before probe separation and by insulation only after separation affords a state-of-the-art solution. For Pioneer probes, heaters may be required after separation as well because power is not available for preheating the probe.

H. REFERENCES

II-1 *Preliminary Model Atmospheres for Jupiter.* JPL Section Document 131-10. December 31, 1969.

II-2 Material on Jupiter Atmospheres dated 8 May 1970 communicated by R. G. Nagler of JPL on 14 October 1970.

II-3 Material on Jupiter Atmospheres dated 8 October 1970 communicated by R. G. Nagler of JPL on 14 October 1970.

II-4 JPL letter 290-RGN:cth from R. G. Nagler, to S. Ducsai dated 28 August 1970.

II-5 *Protection Requirements for the Resistance of Meteoroid Penetration Damage of Interplanetary Spacecraft Systems.* Technical Report No. 32-410. Jet Propulsion Laboratory, Pasadena. July 1, 1964.

II-6 *Space Environment Criteria Guidelines for Use in Space Vehicle Development.* NASA TMX 53957. 1969 Revision, 2nd Edition, August 26, 1970.

II-7 *Titan III/Centaur Family Launch Vehicle Definition for a Jupiter Entry Mission Study.* JPL Section Document 131-09. January 14, 1970.

II-8 *Constants and Related Information for Astrodynamic Calculations.* JPL TR 32-1306. 1968.

II-9 *Outer Planet Spacecraft System Descriptions.* JPL Section Document 131-08. December 31, 1969 and Revision dated June 30, 1970.

II-10 *Brief Summary of DSN Capabilities for the Jupiter Atmospheric Probe Mission.* JPL Document 131-11. March 30, 1970.

II-11 *Tracking and Data Systems Estimated Capabilities for the Mars 1973 Mission.* JPL Document 611-1. December 1, 1968.

II-12 W. G. Melbourne: "Planetary Ephemerides." *Astronautics and Aeronautics.* May 1970.

II-13 J. E. Myers: "Guidance and Control Subsystems Calibration Studies." Martin Marietta Corporation, Denver, Colorado, (Contract NAS1-9000), June 1970.

III. CONCLUSIONS

A. GENERAL

A Jupiter atmospheric entry mission, with adequate date return from a probe obtaining direct science measurements to a pressure of 1000 atmospheres is feasible. The major uncertainty that affects the mission performance is the Jupiter atmospheric model, which defines the characteristics of pressure, temperature, composition, and hence the microwave attenuation. The science objectives identified in the contract statement of work can be satisfied at 10 to 50 atm depth in the nominal model atmosphere and at 100 to 300 atm in the cool-dense model. However, descent to greater depths, for exploration of the unknown, is highly desirable.

The major costs of survival to 1000 atms are in power and weight penalties, constraint of postencounter objectives, thermostructural technology, and system complexity. The power penalties with depth accrue primarily from the increased atmospheric attenuation and the extended mission times. The weight penalties due to increased pressure and temperature are aggravated by the increased power requirement. Communications geometry requirements for the optimum probe/spacecraft relay link constrain the periapsis flyby radius, thereby limiting the achievement of postencounter objectives, such as targeting the spacecraft for a Grand Tour trajectory. The thermostructural technology requirements increase with depth as evidenced in the performance of external insulation and hydrogen compatibility with exposed materials at high pressure and temperature. At depths beyond about 300 atm, the single probe concept reaches excessive total system weights. The more complex, but lighter, split probe dual-relay concept is required for depths from 300 atm to 1000 atm. Depths of less than 300 atm must be accepted for greater flyby radii.

Both the TOPS and the Pioneer F/G spacecraft provide an adequate base for the Jupiter atmospheric entry mission planetary vehicle design. However, the TOPS spacecraft is better suited to a probe mission than is the Pioneer F/G. While the Pioneer can be modified, or the probe design extended to avoid Pioneer modifications, the acquisition of an accurate attitude reference for the deflection maneuver from a spinning spacecraft appears less certain than from a three-axis stabilized spacecraft. TOPS should be able to position the probe within 0.4 deg of any required attitude. Turn accuracy for Pioneer is 7% of the turn angle, which may be 50 deg or greater. In the case of probe missions using Pioneer, a probe attitude control system precesses the probe from its Earth orientation to the deflection attitude. While this appears feasible, it is costly to implement.

The relay link is preferred over the direct link because the relay link provides a much greater capability and versatility and reduced overall system weight. Also, the relay link allows extended atmospheric descent times and therefore, greater probe depths than are possible with the direct link.

The study has shown that a broad range of missions are feasible using a spacecraft relay link. Within this range, potentially viable missions would include a TOPS with probe to 300 atmospheres, a dual-planet probe mission in conjunction with a 1979 Grand Tour to 17 atm, and a probe with expanded science payload to approximately 100 atm, delivered from a 1978 Grand Tour spacecraft.

B. SCIENCE CONCLUSIONS

The Jupiter atmospheric entry mission study evolved from the basic science questions specified in the statement of work. The services of a group of interested planetary scientists (Chapter II.B), were solicited to help interpret the scientific objectives in terms of relevant measurements and instrumental techniques compatible with a descent probe. The conclusions in this section reflect their views as well as those of the Martin Marietta personnel involved.

An entry into either the Equatorial Zones or the North Equatorial Belt is both adequate and desirable for the objectives of a first generation probe. The two regions around 5°N and 5°S probably have low atmospheric turbulence based on long-term observations of Jovian cloud dynamics, and on general principles of atmospheric dynamics (Volume III, App D). Entry near the subsolar point is most desirable for solar flux measurements, but would limit the accomplishment of most other objectives due to the weight penalties. The best compromise is obtained by a shallow (posigrade) entry at about 50 to 60 deg from subsolar to allow sufficient time for the probe to descend below the cloud tops before planet rotation carries it into darkness. A dark side entry is acceptable for a first-generation mission since the most important questions concerning Jupiter can be answered by measurements on either the light or dark side.

There are four identifiable levels of descent indicated by the observables. These are: (1) below the turbopause to determine the H/He ratio and gross atmospheric composition: (2) the 1 to 10 atm level to obtain information on the pressure-temperature profiles and clouds; (3) the 100 to 300 atm levels to satisfy the requirements of most of the observables; and (4) the 500 to 1000 atm levels to satisfy curiosity and for the sake of exploration of the unknown (Volume II, Chapter III). All atmospheric measurements should begin above the clouds at P $\sim$ 0.1 to 0.3 atm in order to locate the cloud tops and to correlate with remote sensing data.

Although descent to 1000 atm would be desirable for exploration of the unknown; it is not required to meet the basic science objectives. Measurements from above the visible cloud tops down to below the base of the expected H_2O cloud are required for an adequate accomplishment of the scientific objectives. Since reduction of present uncertainties in the atmosphere profile and composition is unlikely prior to a probe mission, a descent to at least 100 to 300 atm is required to ensure reaching below the H_2O cloud in the cool-dense model. The optimization of measurement quality down through these levels should take precedence over reaching the 1000 atm levels on a first-generation mission. Because of this, it was concluded that the nominal payload should be expanded and modified to optimize the accomplishment of the stated objectives.

In particular, the capabilities of the GC/MS should be expanded and visual photometers, a nephelometer, IR radiometers, and an RF lightning detector and microphone should replace the nominal photometers (Volume II, Chapter III and Volume III, Chapter II). The reduced, or minipayload, consisting of a reduced capability mass spectrometer, pressure, temperature and nephelometer, has a measurement capability less than the minimum desired by the scientists; however, the minipayload does provide answers to many of the basic questions and its reduced weight makes it possible to deliver two probes from a Grand Tour trajectory, one to Jupiter and one to Uranus or Neptune.

C. TRAJECTORY CONCLUSIONS

Evaluation of typical interplanetary trajectories and launch dates has shown that a variety of mission options are available in conjunction with the Jupiter atmospheric entry mission and that the launch and encounter choices present no major constraints on the mission selection. Targeting and relay link communications geometry are the major mission design considerations.

Both targeting and payload capabilities are influenced by the trajectory and mission year. It was concluded that for the 1978 launch year, Type II trajectories provide about 25% more payload weight than Type I trajectories, such as Grand Tour. Also, the Type II trajectories result in light side missions at low entry angles while the Type I trajectories result in dark side missions at equivalent entry angles. The lower entry angles result in lower deflection velocity requirements and lower entry heat shield/ aeroshell weight penalties. On the other hand, Type II mission durations are about 18 months longer than comparable Type I missions.

Targeting is greatly influenced by the data link concept; e.g. for a reasonable direct communication link between the probe and Earth, entry should occur within 25 deg of the subearth point. Entry angles to achieve this condition approach -50 deg for Type II paths, and -90 deg for Type I paths. The relay link allows targeting to entry angles between about -10 deg and -30 deg for most missions.

The encounter geometry is a major factor in the design of the relay communications link. A major mission objective is the synchronization of the probe's position in the planet's rotating atmosphere with the flyby spacecraft position. It was concluded that for a wide range of descent times and depths, a spacecraft flyby radius of about 2 R_J was optimum and minimized the combined losses due to probe aspect angle, range, and atmospheric attenuation. This optimized trajectory places the probe directly under the spacecraft at flyby periapsis. Postencounter objectives (e.g., Grand Tour) may dictate other flyby radii and consequently reduced descent mission performance.

D. NAVIGATION AND ACCURACY CONCLUSIONS

The error sources considered in the accuracy analysis included those associated with planet ephemeris, the DSN tracking capability, the deflection implementation errors, the uncertainty in the Jovian gravitational constant, and the uncertainty in the 1 atm level of the Jovian atmosphere.

The analysis indicates that the probe deflection implementation errors are the dominant source of dispersions when considering the projected DSN capability and ephemeris accuracy, and that the navigation errors (e.g., relative location of the planet and spacecraft) have little effect on the accuracy of the missions considered. Thus, onboard sensors that further reduce these navigation errors contribute very little. The uncertainty in gravitational constant can also be neglected. However, the uncertainty in the 1 atm pressure level must be considered for entry angle dispersion at the very shallow entry angles near -10 deg. Above about -15 deg γ_E this error can also be neglected.

The dominant error source, the deflection maneuver, results in dispersions in targeting, entry angle, and lead time in the probe/spacecraft geometry. For a nominal entry angle of -20 deg and a flyby periapsis of 2 R_J, the targeting errors with a deflection system capability (3σ) no greater than 1% proportionality and 1.5 deg pointing will be ±0.75 deg crossrange and ±3 deg downrange. These are sufficiently small to permit achievement of a desired target area. For the same deflection system, entry angle errors will be ±1.52 deg. The only error that significantly affects the system design is the lead time uncertainty of ±17.6 minutes. This results in probe aspect angle uncertainties of 26 deg and requires increased transmitter powers.

E. PROBE SYSTEMS ENGINEERING CONCLUSIONS

It is concluded that atmospheric probe systems can be designed to survive the Jovian entry and descent to 1000 atm pressure and are feasible. With the exception of the heat shield, existing technology with some moderate additional development is adequate. Although the assumed heat shield weight fractions of 30% to 40% have a large impact on the system weight, the study showed that a 50% increase in the heat shield weight fractions could be accommodated without invalidating the feasibility of the designs. Such a variation results in an overall probe system weight increase of 30%. In addition to the heat shield, the other technologies critical to mission feasibility are the external insulation/pressure vessel performance, and the materials' compatibility with hydrogen at extreme pressure and temperature. Long-lead development items include high-power solid-state amplifiers, probe deflection maneuver systems development, remotely activated batteries, and components for the high deceleration entry environment. In certain other areas, including pressure vessel seals, feedthroughs, and protection for external optical surfaces, development is required.

The parametric study showed probe system design and weight sensitivities to the basic mission parameters. The sensitivity to entry angle is quite pronounced, due primarily to the influence of increasing dynamic pressure on the aeroshell structure weight. The impact on probe weight is aggravated by the added heat shield weight required although the heat shield weight fraction itself does not increase significantly with increasing entry angle. The probe system weights are nearly doubled when going from an entry angle of -20 to -50 deg. The radius of periapsis weight penalty effect is greater for single probes than split probes since the transmitter power increase with increasing range depends on the depth of the probe in the atmosphere and the upper part of the split probe stays at shallow depths. The transmitter power limit of 40 watts limits the single probes to depths of about 300 atm while the split probes can attain 1000 atm. Science payload weight has a relatively mild effect since almost the same communications package is required for the different science payloads and it comprises the largest portion of the descent probe volume. In the range of 40 bps, bit rate effect on total system weight is likewise not pronounced because its influence on transmitter power is diminished by the carrier requirements. A limiting value in terms of reasonable transmitter power is reached with increasing bit rate before weight penalties become large.

Designing for both the nominal and cool-dense atmosphere models results in a 30% increase in total system weight for a 300 atm single probe with a -20 deg γ_E. However, if the probe is designed for the science criteria of reaching below the clouds in the cool-dense (100 to 300 atm) and in the nominal atmosphere (10 to 50 atm), then no weight penalty results.

For the descent probe, external insulation is required on pressure shells at large depths to protect the pressure shell to working temperatures of about 1100°F for most candidate materials. In addition, multilayer internal insulation and phase-change material provide adequate protection for the internal equipment at the extreme depths.

F. DATA RETURN CONCLUSIONS

It is concluded that data return systems can be designed with existing technology to provide the required mission data rates at sufficient signal strengths to overcome the Jupiter atmospheric attenuation, the range losses to the spacecraft or Earth, and the losses associated with excursions in the probe antenna aspect angles relative to the receiver.

The atmospheric attenuation characteristics were estimated based on the Jupiter atmosphere models; therefore, the uncertainties in the attenuation results are at least as great as those of the atmospheric models themselves. The estimated zenith attenuation is as high as 36 db at 1000 atm pressure at the DSN frequency of 2.3 GHz. Lower frequencies reduce the losses, particularly at the greater depths. However, background noise rises sharply with decreasing frequency as does antenna size and weight. This gives an optimum frequency that ranges from around 0.85 GHz for deep penetrations approaching 1000 atm to 2.3 GHz for shallow penetrations of around 15 atm or less. For the cool-dense atmospheric model with considerably higher attenuation than the nominal model, a reduction in frequency of about 30% is required for optimum design. At depths greater than 300 atm where the attenuation becomes increasingly severe, the use of the split probe concept allows optimization of the data return system frequencies and greatly reduces the transmitter power requirements. The lower-to-upper probe link requires only 2 watts of power due to the short range, and the upper probe to spacecraft link requires only from 10 to 20 watts of power because of very little atmospheric attenuation. However, the system is necessarily more complex.

For a relay link, the combined losses due to atmospheric attenuation, range, and probe aspect angle can be minimized by using a flyby periapsis radius of about 2 R_J. This results in the synchronization of the probes position in the planet's rotating atmosphere with the position of the flyby spacecraft.

From this study, certain basic conclusions can be made concerning the data systems. A relay link gives more mission flexibility and higher total data flow with overall lower total weight than a direct link. Search and acquisition modes for frequency lock-on and antenna tracking are required for spacecraft telecommunications relay systems for all missions studied. There is a need for transmitters with output power up to 40 watts at a frequency near 1.2 GHz only if serious consideration is given to exploring atmospheric depths much greater than 300 atm; otherwise, 10 to 20 watt solid-state transmitters are adequate for shallower missions assuming that somewhat optimum trajectories may be selected for the flyby spacecraft. The preferred modulation for a relay link consists of coherent PCM/PSK/PM with convolutional encoding and Viterbi algorithm decoding.

A direct link probe requires a retrodirective array to give meaningful bit rates for any reasonable period of time. This mode requires excessive weight when compared to a relay mode and is restricted to reduced mission duration and fewer total bits of data.

G. SPACECRAFT INTERFACE CONCLUSIONS

It is concluded that integration of the probe system with either the TOPS or Pioneer F/G spacecraft is feasible and that either of these spacecraft can be modified to provide adequate support of the probe mission.

The requirements for support of a probe mission were found to be well within the capabilities of the TOPS spacecraft. It is well suited, technically, to a probe mission.

The Pioneer presents several problems. To avoid serious constraints on the time, and hence depth, of a probe mission, a despun antenna is needed. Depending on the approach taken, this can either aid or aggravate a marginal inertia ratio problem.

One approach involves the use of a partial range tracking antenna which moves the probe mounting position out from the spacecraft center of gravity, thereby increasing transverse axis inertias. These inertias become nearly as large as the roll inertia, thereby resulting in a marginally stable vehicle. While potentially more complex, the boom-mounted antenna provides much broader coverage and offsets the probe induced transverse inertia buildup problem because it is not despun until the probe is ejected.

Depending on the spacecraft science data profile, added data storage provisions would be required for pioneer, or an increase in transmitted power to increase bit rate. Most significant, when off the Earth orientation, the spacecraft has insufficient orientation accuracy to establish probe deflection orientation.

These problems can be overcome by modifications to the probe and to the spacecraft; however, they do add complexity and cost.

H. SAMPLE MISSION CONCLUSIONS

The d sign example mission demonstrates a probe-optimized mission using a TOPS spacecraft at a flyby radius of 2.0 R_J. The resulting entry probe system weight of 430 lb met all requirements in the nominal atmosphere. The cool-dense atmosphere was found to increase the total probe system weight by 30%, which is within the capability of the 7-segment Titan IIID/Centaur/Burner II launch vehicle.

A series of sample missions (A, D, and E) showed use of a Jupiter entry probe with JUN Grand Tour missions feasible. As a group, these sample missions showed the strong interdependence between the Jupiter probe mission and the flyby radius of the spacecraft, hence the JUN Grand Tour trajectory constraints.

- Sample Mission A showed it possible for a single probe to go to 10 atm with a 1979 launch, $R_p = 6.8\ R_J$, and a slightly larger than nominal payload;
- An option of Sample Mission A showed a split probe to 100 atm to be slightly over the limit of launch weight availability;
- Sample Mission D (also 1979) demonstrated the feasibility of a miniprobe to Jupiter and another miniprobe to a subsequent planet. The entry weight of 150 lb allows answering the basic, but not the complex, science questions and in the nominal atmosphere only;
- Sample Mission E (1978) with the smaller $\left(1.6\ R_J\right)$ flyby radius constrains the aspect angles and mission descent time; however, an expanded science payload and a depth of 100 atms is possible.

Sample Missions B and C with identical entry probes, demonstrated the use of the Pioneer F/G spacecraft and also the impact of an orbiter as opposed to a flyby spacecraft. The entry probe mission was unaffected; however, the orbiter mission required a 7-segment Titan as opposed to 5 segments for the flyby due to the added weight for the orbiter mode. Although this mission was limited in depth, its expanded science payload provided a relatively high science value compared to that of the 300 atm design example at nearly equal weights.

Sample Mission F examined a direct link from probe-to-Earth concept. This probe required a reduced payload (similar to Mission D) and a reduced data return rate. The resulting probe entry weight was 500 lb and it attained a depth equal to 17 atm. A large steerable phased array antenna contributes significantly to the system weight. It was concluded that for comparable overall system weights, the relay link has a greatly superior data return capability over the direct link and provides versatility in choosing mission options.

I. MAJOR UNCERTAINTIES

The major uncertainties of this study are discussed in this section.

1. Jupiter Atmospheric Model

The Jupiter atmospheric model uncertainties have a major effect on the required engineering design margins. Designing to the cool dense atmosphere results in large weight penalties in the entry, descent, and communications systems.

The atmospheric uncertainties having the most impact on the mission design and capabilities are:

1) The pressure-temperature structure ($600°K \leq T \leq 1850°K$ at 1000 atm) in the lower atmosphere;
2) The H_2O and NH_3 abundance;
3) The scale heights in the upper atmosphere;
4) The magnitude of winds and turbulence in the lower atmosphere.

With the possible exception of scale height in the upper atmosphere it appears that little can be done to reduce these uncertainties prior to a probe mission. Pioneer F/G occultation experiments may provide information on the upper atmosphere structure.

The atmospheric and cloud composition uncertainties affect the predicted microwave attenuation characteristics. Also, the attenuation prediction techniques require experimental substantiation.

The atmospheric turbulence levels could adversely affect the probe antenna pointing and orientation resulting in further weight penalties or loss of performance.

2. Heat Shield Performance Weight Factors

The heat shield weight fractions were specified by JPL to constrain this study. However, there is considerable uncertainty in the estimated level of the entry heating rates that will reach the heat shield surface and in the performance of the heat shield material at the extreme heat rate and dynamic pressure conditions that are anticipated.

3. Science Question Priorities

In this study, all science questions were assumed to have equal priority. The assignment of different priorities to the questions will have an effect on the engineering implementation and on the required targeting, depth of penetration, and descent mission time required to obtain the measurements. The optimum descent mission profiles and science instrument payloads are a direct function of the science question priority. Variations in mission weight limits would result in variations in the optimum science payload complement and most effective mission depth.

4. Deflection Implementation Accuracy

A major uncertainty with respect to accomplishment of the mission objectives is associated with deflection of the probe from the approach trajectory to the entry corridor. Accuracy analyses have demonstrated that the errors of navigation during interplanetary cruise will be small when compared to the anticipated errors of deflection and coast to the planet. Contributors to the latter include the tolerances of design, build, and alignment, plus the ability to control probe dynamics. It appears mandatory to include an active attitude control system in the probe or to target the planetary vehicle toward the planet and then deflect the spacecraft away to its flyby trajectory. A capability comparable to that of the Viking Lander system seems necessary and adequate.

Closely related to uncertainty in the deflection maneuver is the relation of the entry trajectory of the probe to the flyby trajectory of the spacecraft. The communications geometry has been shown to play a very important part in mission accomplishment and the dominant factors in the geometry are the deflection velocity (magnitude and direction), the flyby periapsis radius of the spacecraft, and the lead time between the probe entry and spacecraft periapsis passage. This problem is magnified as the freedom to select optimum values is diminished, as in coupling the entry mission to a Grand Tour Mission.

5. Insulation Performance

For the probes that descend to 300 to 1000 atm, the optimum heat protection system design called for external insulation to protect the pressure vessel from the full ambient temperature. The performance capability of the external insulation at the extreme ambient pressure and temperature is a major design uncertainty which will require testing and development in the simulated environment.

6. Materials Compatibility with the Hydrogen Environment

There is some uncertainty in the compatibility of various materials with hydrogen at very high temperature and pressure. Although the better known phenomenon of hydrogen embrittlement of metals does not appear applicable to conditions encountered in this study, it is necessary to obtain experimental data to assure materials performance.

IV. RECOMMENDATIONS

A. RECOMMENDED ENGINEERING R&D

The study has identified areas in which development work should be pursued as part of the planning for a Jupiter mission. The following research and development tasks are recommended:

1) Development of heat shield designs, materials, and processes of fabrication supported by testing after exposure to the space environment (Volume II, Chapter II, Section A and Chapter V, Section I). (This development represents the major uncertainty identified by the study.)
2) Development of improved technology for aeroshell structure at high dynamic pressures where steep entry angles (e.g., light side entry) may be required (Volume II, Chapter V, Section E.2).
3) Development of models for attenuation of microwaves by the Jupiter atmosphere supported by laboratory test verification (Volume II, Chapter V, Section J).
4) Further evaluation of the navigation and deflection implementation accuracy problem (Volume II, Chapter V, Section H).
5) Development of external insulation materials and other thermal control materials and processes of installation supported by testing after exposure to the space environment (Volume III, Chapter V, Section A.3a).
6) Development of 40 watt solid-state phase modulated transmitters (Volume II, Chapter V, Section D.3c; and Volume III, Chapter IV, Section I.2).

7) Development of reliable batteries capable of remote activation after long storage life (Volume III, Chapter IV, Section I.3).
8) Selection or development of materials for compatibility with hydrogen at high temperature and pressure (Volume III, Chapter V, Section B.1).
9) Verification of long storage life reliability for all components contemplated for the Jupiter mission (Volume II, Chapter VIII).
10) Verification of capability of withstanding high deceleration loads for all components contemplated for the Jupiter mission (Volume II, Tables VI C-1, VII A2-1, -B2-1, -D2-1, -E2-1).
11) Development of materials and methods of forming seals and penetrations compatible with the Jupiter environment after exposure to the space environment (Volume III, Chapter V, Sections A-3a and B-1a).
12) Development of positive methods of providing stability for a descending sphere (Volume III, Chapter V, Section D-1).

B. RECOMMENDED SCIENCE INSTRUMENT R&D

All of the instruments will require development to perform properly through the lower atmosphere and clouds on Jupiter. The most critical requirements are for an early start on the development of:

1) A suitable high-pressure, high-temperature gas sampling and pressure reduction system for the gas chromatograph/mass spectrometer;
2) A gas chromatograph/mass spectrometer capable of providing isotopic analyses and separations of H, H_2, D, H_3, HD, He^3, D_2, and He^4 while also covering the range up to m/e 60;
3) A cloud particle collecting and processing system and suitable gas chromatograph columns for rapid separation and identification of complex molecules.

Other instrumental problems requiring further study or development include:

1) A method of preventing condensibles from blocking pressure inlet and sampling ports;
2) IR transparent windows capable of withstanding high pressures at high temperatures;
3) A method for preventing or removing or compensating for condensation on optical windows (Photometers, radiometers, nephelometer);
4) A technique for determining the temperature gradient very precisely (±0.02° K/km).

(Volume II, Chapter III, Section D and Volume III, Chapter II, Section C)

C. RECOMMENDED FUTURE STUDIES

An in-depth mission study is required to refine and optimize the mission design and operation and to optimize the system and subsystem designs.

The science consultants indicated a preference for small probes to two planets over two small probes to one planet. However, they also indicated that a large single probe to Jupiter with an expanded science payload was more desirable than two small probes to Jupiter. This leads to a study to evaluate and compare these two basic options:

1) Grand Tour swingby with probes to multiple planets. This mission provides relative ease of implementation and cost savings by combining probe and Grand Tour programs. A basic trade of optimum science payload and depth for the two planets would maximize the overall science value.
2) Probe optimized for Jupiter only. Again the basic trade is optimum science payload versus depth. In this case lightside targeting is another important variable with respect to science value. This mission represents the best possible science mission capability for a single planet.

In addition, the scientists desire a complementary spacecraft science package for either flyby or orbiter missions that would include such items as TV and IR coverage of the entry site. The overall mission science evaluation should integrate both the spacecraft and probe science capabilities.